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TECHNICAL REPORT: COMPUTER PROGRAM DEVELOPMENT AND METHODOLOGY

SATURN IN-FLIGHT EXPERIMENTAL
PAYLOAD STUDY

CONTRACT NAS8-20236

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F O R E W O R D

This document, Volume III of the report on the Saturn In-Flight Experimental Payload Study, contains that portion of the Technical Report pertaining to the Computer Program Development and Methodology. The study was conducted by the Fort Worth Division of General Dynamics for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under Contract NAS8-20236. The study was established by the Advanced Systems Office of NASA-MSFC as part of an effort to provide for the orderly and economic utilization of space vehicle hardware in the tasks devoted to the accumulation of scientific data.

The complete results of this study are documented in four volumes:

Volume I - Summary

Volume II - Technical Report: Design of In-Flight Experiments

Volume III - Technical Report: Computer Program Development and Methodology

Volume IV - Utilization Instructions.

This study was performed during the period beginning July 1965 and ending February 1966. The general guidelines of the study were set forth by NASA-MSFC in RFQ DCN 1-5-23-00009-01 and RFQ DCN 1-5-23-00010-01, and the Fort Worth Division has based the study effort on these guidelines in order to obtain the results described herein.

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SECTION 1

INTRODUCTION

1.1 GENERAL

This document is Volume III of the report on the Saturn In-Flight Experimental Payload Study. It contains that portion of the Technical Report pertaining to Computer Program Development and Methodology. The remaining portion of the Technical Report pertains to Design of In-Flight Experiments and constitutes Volume II. The study was performed by the Fort Worth Division of General Dynamics for the George C. Marshall Space Flight Center.

By the utilization of the secondary payload capability of the Saturn family of launch vehicles, NASA can provide an efficient means for conducting the large number of Earth-orbital experiments that has been suggested by the scientific community. Since it is to be assumed that the mission of each launch vehicle is designed to attain specific objectives associated with the primary payload only, it is essential that the in-flight experiments and the launch vehicle be properly mated to provide for efficient utilization of the remaining mass and volume capability of the launch vehicle and for the accomplishment of a high percentage of the experiment data acquisition objectives. Because of the combination of numerous vehicles with varying missions and capabilities and a large number of experiments with varying requirements, the evaluation of the vehicle/experiment mating presents a significant management problem. The basic objective of the Saturn In-Flight Experimental Payload Study is to provide NASA with a management tool in the form of a computer program which can be used to make a rapid evaluation of numerous potentially attractive space experiments that constitute possible secondary in-flight payloads for the Saturn family of launch vehicles.

1.2 APPROACH

To attain this overall study objective, two major study tasks were specified: (1) an analysis of the physical characteristics of sensors and associated equipments for use as possible experimental payloads on Saturn-class vehicles and the mission effectiveness values of these experiments as a function of the initial elements and/or mission parameters of the deployed orbit and (2) the development of a computerized methodology for the technical evaluation and rating of these potential in-flight experimental payloads.

The technical approach used throughout the study is based on the development of Program SEPTER (Saturn Experimental Payload Technial

Evaluation and Rating). Two fundamental criteria are employed in Program SEPTER to evaluate the experiments that are being considered for possible inclusion on a Saturn flight: (1) physical compatibility of the experiments with possible locations aboard the vehicle, and (2) experiment/mission effectiveness. Experiment/mission effectiveness is defined as the percent of the data acquisition objectives which would be attained by including a particular experiment on a given Saturn flight. The physical compatibility of an experiment package with a vehicle location refers, in this study, not only to mass/volume compatibility but also to compatibility with the thermal, acoustic, vibration, and electromagnetic environments.

1.3 COMPUTER PROGRAM - SEPTER

The overall structure and the key concepts of Program SEPTER are shown in Figure 1-1. This program contains provisions for operating in two basic modes. In the Mode I operation, the compatibility and effectiveness of single experiments are determined. In the Mode II operation, the arrangement configurations and compatibility of multiple experiments are analyzed, and desirable arrangements are determined.

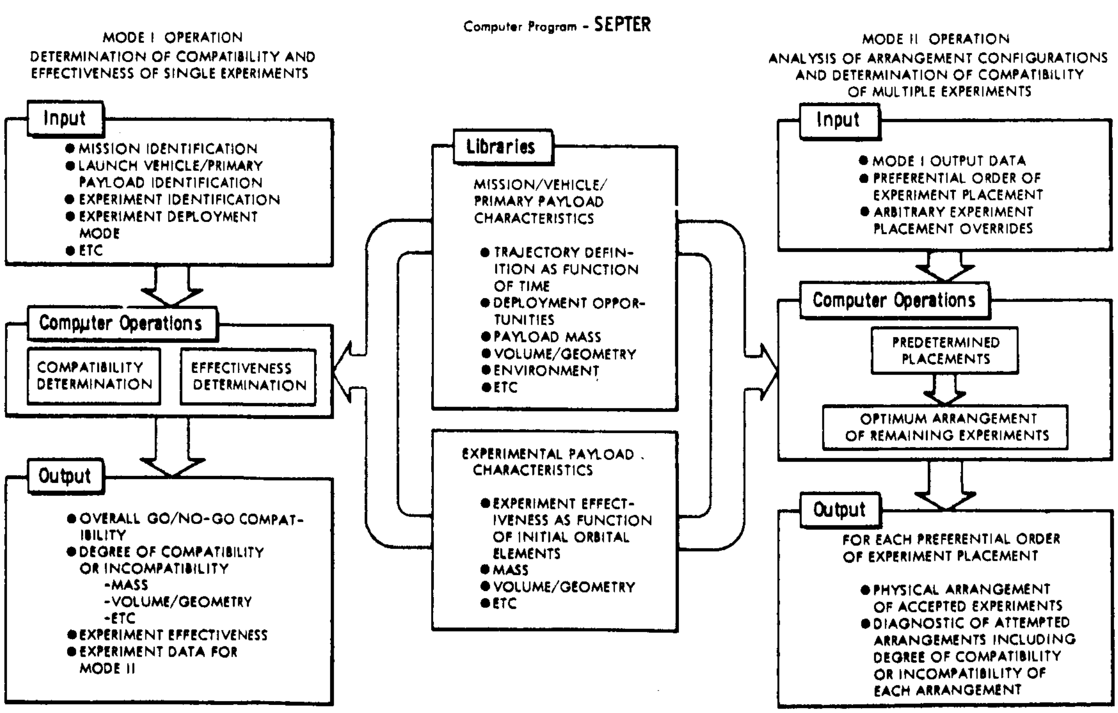


Figure 1-1 SATURN EXPERIMENTAL PAYLOAD TECHNICAL EVALUATION AND RATING

Data used in the single experiment or Mode I analysis consist of mission, launch vehicle/primary payload, and experiment identifications and associated information, such as experiment deployment mode, etc. These identifications will result in selections from the libraries of the mission profile, the potential experimental payload locations aboard the vehicle, and the experiments to be considered in the particular "mission." The program is then used to compare the libraries of mission/vehicle/primary payload characteristics and experiment characteristics with these data and to determine the compatibility and effectiveness of each individual experiment. The Mode I output consists of a listing of the experiments, along with information on their individual overall GO/NO-GO compatibility, degree of compatibility or incompatibility, and effectiveness.

After an examination of Mode I output, NASA management will establish the desired order in which the experiments are to be loaded aboard the vehicle and formulate a preference list.

The data used in the multiple experiment or Mode II operation consists of a preference list, a compatibility library from Mode I output, problem control data, and library overrides. The Mode II output is in the form of printed results in which the accepted experimental payloads from the preference list and the cavities within which they have been placed according to the predetermined and optimal arrangement analyses are listed.

1.4 PROGRAM PLAN

The basic program plan shown in Figure 1-2 was developed by the Fort Worth Division of General Dynamics in order to achieve the objectives established for this study. The use of this approach permits (1) an analysis of the physical characteristics and mission sensitivity of experiments of in-flight payloads for Saturn-class vehicles and (2) the determination of a computer methodology for the technical evaluation and rating of these in-flight experimental payloads. The technical plan is divided into the individual study areas associated with the experiments-related task (Task I) and the computer methodology development task (Task II).

The Task I studies were devoted to (1) a thorough definition of the objectives, data acquisition requirements, and sensors for each of a group of representative experiments; (2) establishment of the physical characteristics of each individual experiment by synthesizing self-contained experiment packages on the basis of sensor requirements; (3) analysis of experiment effectiveness variations as a function of experiment-deployment orbital elements; and (4) computer mechanization of the computation of effectiveness values and physical characteristics.

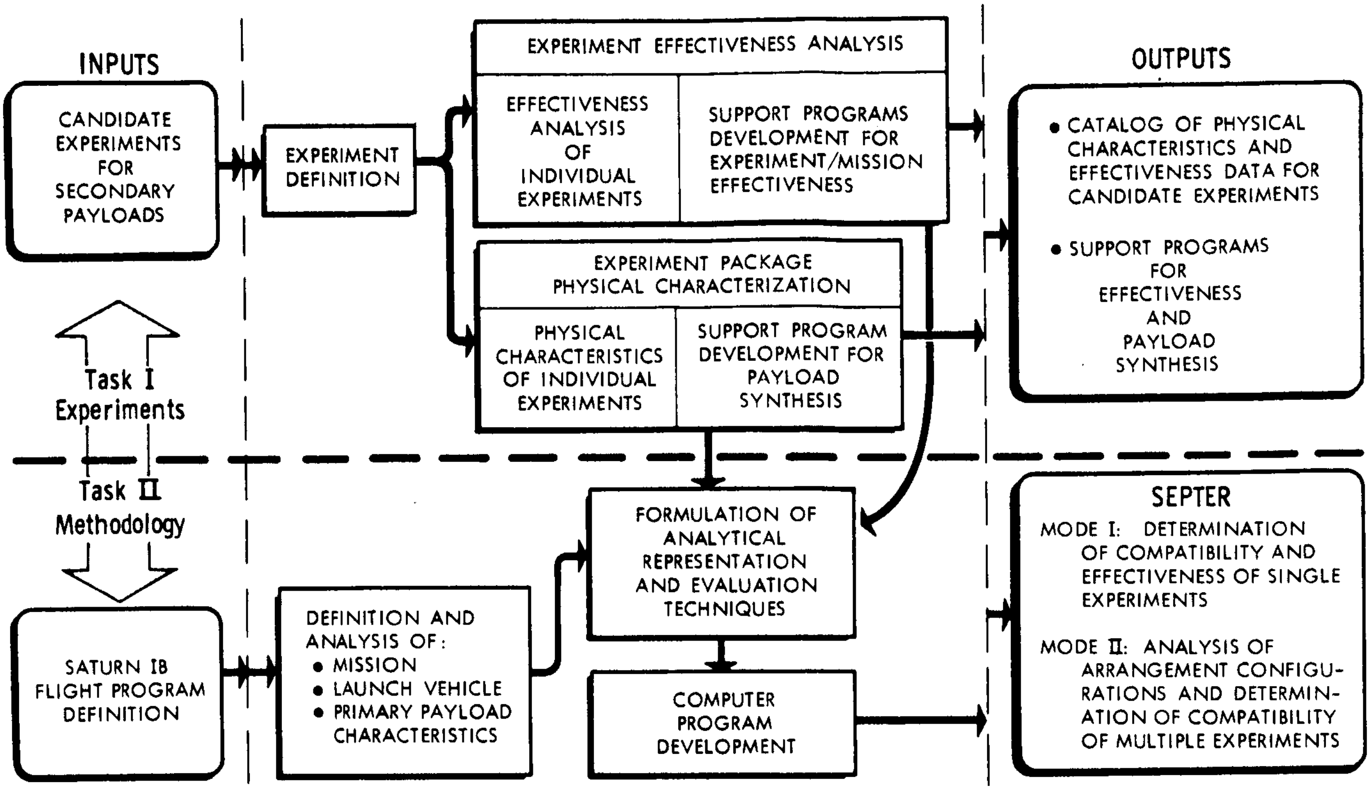


Figure 1-2 BASIC PROGRAM PLAN

The Task II studies were devoted to (1) a definition and analysis of the relevant mission, vehicle, and primary payload characteristics of the vehicle configurations to be considered; (2) development of analytical representations to be used in the computer program to evaluate and rate single-experiment compatibility/effectiveness and to analyze multiple arrangement/compatibility; and (3) formulation of the computer program logic, the input and output data requirements and formats, the library data formats, and the options and modes of operation which, when combined with the analytical representations, will yield the operable computer program.

1.5 GUIDELINES AND GROUND RULES

A number of guidelines and ground rules were specified at the beginning of the study in order to establish the overall study philosophy and to limit the scope of the experiment and vehicle analyses. The experiments considered in this study constitute secondary payloads in that the missions on which these experiments may be flown have been designed to attain specified objectives associated with the primary payload. For example, the primary missions which were used in the mission characteristics library of the computer program are the Saturn

IB/Apollo flight test missions. The basic Apollo spacecraft (Command Module, Service Module, and Lunar Excursion Module) is the primary payload, and any additional experimental packages carried on these flights would then be secondary payloads. Although other vehicle configurations should eventually be included in the launch vehicle/primary payload characteristics library, the Saturn IB/Apollo - including the Command Module, the Service Modules, and the Lunar Excursion Module - was chosen as the baseline configuration for this study.

The Fort Worth Division of General Dynamics acknowledges the prerogative and responsibility of NASA to define and approve in-flight experiments. However, in order to understand how the computer methodology may be affected by differences in (1) the physical characteristics of experiment packages and vehicle cavity locations and (2) the requirements for realistic examples of experiment effectiveness, it was necessary for the Fort Worth Division to define a number of potentially attractive in-flight experiments. In establishing configuration designs for these experiment packages, primary emphasis was placed on self-contained packages; that is, consideration was not given to using the support capabilities of on-board equipment or to the possibility of sharing subsystems among experiments. The analysis of physical compatibility was basically performed by considering completely self-contained packages; however, certain vehicle-dependent packages, which are self-contained packages exclusive of power and communications subsystems, were also considered. The experiment packages were designed to assure that they do not in any way interfere with the primary payload. Futhermore, the package designs were based on the assumption that only a minimum of astronaut participation will be allowed, i.e., only to effect off-on switching, film retrieval, etc.

1.6 SUMMARY OF MAJOR STUDY ACCOMPLISHMENTS

The major tasks which have been accomplished as a result of this study effort are summarized below.

1. From the list of 85 experiments provided in <u>NASA Experiment Descriptions for Extended Apollo Earth-Orbit Flights</u>, 30 experiments were selected which were representative of the list and were compatible with the study ground rules. The following were accomplished in the case of each of the 30 experiments:

   a. The physical characteristics of the experiment sensors and the ancillary systems -- attitude control, data automation, communications, electric power, and thermal control -- were defined.

b. The thermal, vibration, acoustic, and electromagnetic environmental requirements were established.

c. Conceptual design drawings were prepared, and the mass, volume, and geometry of the experiment were determined.

d. The deployment requirements were defined.

e. Preliminary reliability, development schedule, and cost analyses were performed.

2. The pertinent mission characteristics (trajectory parameters, sequence-of-events, and experimental payload possible deployment modes) of a typical Saturn IB/Apollo Earth-orbital mission were defined and analyzed.

3. A total of 53 cavities (potential payload locations) were identified on the Saturn IB/Apollo vehicle. The following were accomplished in the case of each of the 53 cavities:

   a. Isometric drawings were prepared showing the cavity shape and volume.

   b. The mass capacity was determined.

   c. The thermal, vibration, acoustic, and electromagnetic environments were established.

   d. The deployment capability was defined.

4. A methodology was developed for describing experiment and cavity volume/geometry by the use of standard geometric shapes (sphere, cylinder, and parallelepiped). Each experiment was represented by its total volume and standard shape of its critical component. Each cavity was defined by its total volume and by its capacity to accommodate the standard shapes.

5. A methodology for describing experiment effectiveness as a function of the initial elements and/or mission parameters of the deployed orbit was developed, and parametric effectiveness analyses were performed on example experiments.

6. A computer program (SEPTER) was developed to evaluate and rate in-flight experimental payloads. The overall capabilities of this program are a result of the development of some unique and simplified methodologies which are reasonably accurate for the solution of generally complex problems. These methodologies include the following:

   a. The simulation of experimental payload deployment modes and the calculation of the orbital elements and/or mission parameters for the deployed orbit.

   b. The computation of experiment/mission effectiveness as a function of the initial orbital elements of the deployed orbit. A technique was developed in which three types of effectiveness factor relationships are utilized: (1) continuous function of two variables, (2) step function of two variables, and (3) continuous or step function of one variable. Two interpolation techniques are available.

   c. The determination of the experimental payload-mission/vehicle compatibility with numerous physical and operational criteria. A reasonably simple technique was developed for the determination of geometric compatibility between arbitrarily shaped cavities and experimental payloads represented by standard shapes.

   d. The determination of multiple experimental payload arrangements aboard a vehicle. A technique was developed which satisfies all constraints and can be used directly to search for a non-unique "optimal" arrangement.

7. A computer program (DESIGN) for determining limited physical characteristics of arbitray experiments was developed as a support program for Program SEPTER. DESIGN replaces the manual subsystem synthesis tasks of designing experimental payloads and provides "first-pass" estimates of mass and volume requirements.

part I

EXTERNAL ANALYSIS

SECTION 2

DEFINITION AND ANALYSIS OF MISSION CHARACTERISTICS

2.1 GENERAL

A preliminary task in the overall development of Program SEPTER was to define and analyze the pertinent mission characteristics of individual Saturn missions. This task was required in order (1) to obtain representative data for the Mission/Vehicle/Primary Payload Characteristics Library and (2) to formulate the progrom logic for representative deployment modes and the calculation of mission parameters and orbital elements for any specified deployment mode and deployment time.

2.2 SATURN MISSION TYPES

Initially, a survey of Saturn missions was made to determine which mission types should be considered for definition and analysis. The Saturn IB/Apollo vehicle/payload was used as the basic configuration for the study, and missions compatible with this configuration were sought. The Saturn IB flight program was investigated in particular. Mission objectives and plans were obtained from Reference 2-1. Nominal trajectory data were obtained from References 2-2 through 2-6. These early scheduled missions were found to include suborbital and Earth-orbital types of missions and various payload configurations.

The missions that were found to be compatible with the Saturn IB/Apollo configurations and development program are the Earth-orbital, low-altitude, low-inclination type. Representative launch trajectory data for this mission type were obtained from Reference 2-6. These data are included in the Mission/Vehicle/Primary Payload Characteristics library of Program SEPTER.

Other mission types, such as the suborbital missions, were investigated for possible inclusion in the program. Although missions of this type are not precluded by the program, their use for in-flight experimental payloads is considered to be limited because of factors such as (1) short time duration of these missions and (2) mission and vehicle physical constraints on experimental payload ejection.

2.3 PERTINENT MISSION CHARACTERISTICS

The mission characteristics that were found to be pertinent to the overall development of the computer program may be categorized as data of the following types: (1) trajectory parameters, (2) sequence-of-events, and (3) experimental payload deployment opportunities/constraints and possible modes.

2.3.1 Trajectory Parameters

Time histories of the trajectory parameters of a typical Saturn IB/Apollo launch trajectory were obtained (for operational vehicle SA-207) from the data given in Reference 2-6. Time histories of the parameters which are used to define the Earth-relative position (latitude, longitude, and altitude) of the vehicle and its inertial velocity vector (veloctiy magnitude, flight path angle, and azimuth angle) are given in Figure 2-1. These six position and velocity parameters completely specify the vehicle's orbital elements at a given time. They are also used for the determination of the orbital elements of an experimental payload for any deployment mode and deployment time.

2.3.2 Sequence-of-Events

Mission sequence-of-events data are required to identify (1) experimental payload deployment opportunities/constraints and possible modes and (2) physical environments to which the experiments are subjected during various mission phases. The staging, jettisoning of hardware, separation of the payload from the vehicle and the separation, transposition, and docking maneuvers of payload components are typical events which must be defined as a function of time in the mission.

Typical sequence-of-events prior to injection of the primary payload are depicted along with the launch trajectory data in Figure 2-1. Data representative of the sequence-of-events subsequent to primary payload injection into orbit are given in Table 2-1 for a manned Apollo development mission. These data are approximate in that scheduled mission data for the orbital phase were not available. The data given in the table were formulated primarily to ensure the compatibility of computer program logic with numerous possible orbital maneuvers and to provide representative data for the Mission/Vehicle/Primary Payload Characteristics Library.

2.3.3 Experiment Deployment

Mission characteristics defined and analyzed for the computer program were mission imposed deployment opportunities/constraints and possible deployment modes as a function of time in the mission. A secondary analysis was conducted to determine the effects of applying small propulsive velocity increments to the experimental payload at deployment time (for experimental payloads which require ejection). The objective of this analysis was to provide data to be used in establishing propulsion requirements with which to achieve experimental payload orbits more compatible with data acquisition objectives, thus increasing experiment effectiveness.

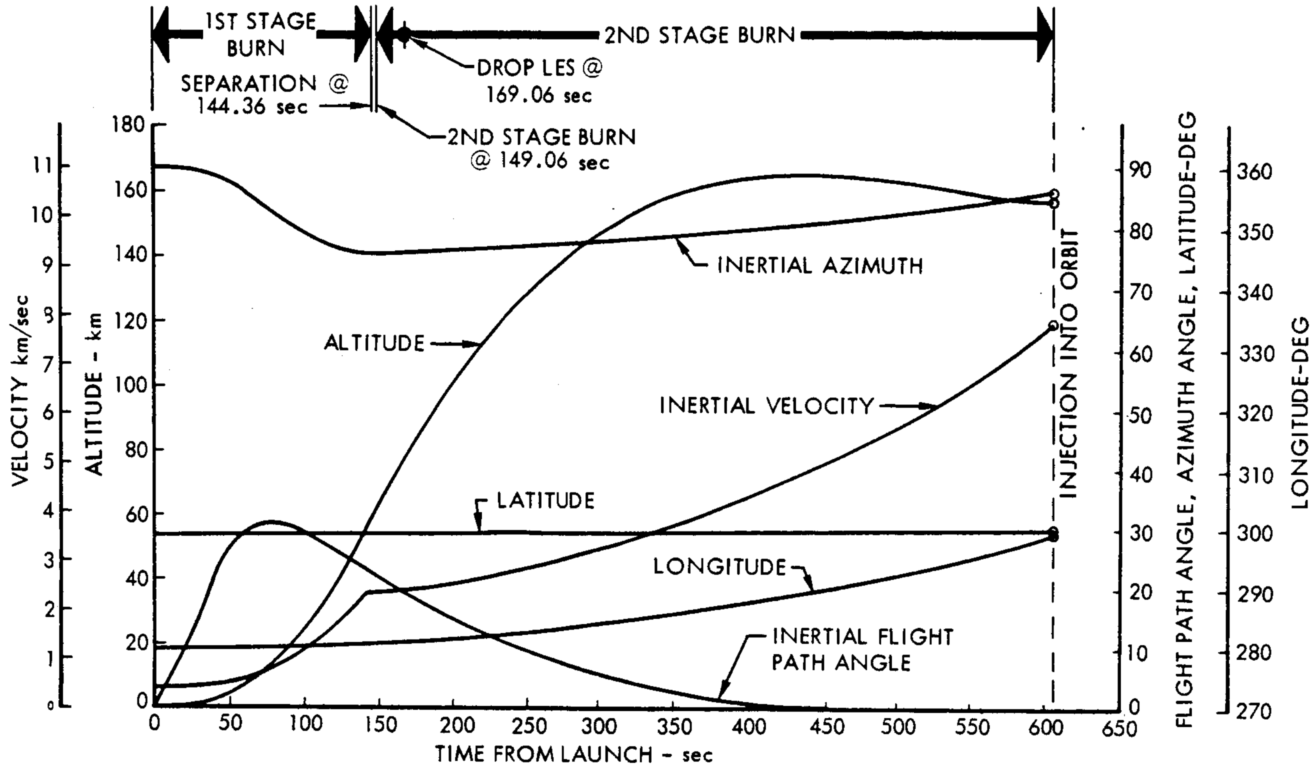


Figure 2-1 SATURN IB TYPICAL LAUNCH TRAJECTORY TIME HISTORY

TABLE 2-1

TYPICAL SEQUENCE-OF-EVENTS AFTER INJECTION INTO ORBIT

REFERENCE ORBIT: APOGEE - 215.8KM PERIGEE - 155.8KM INCLINATION - 30°

Orbital Operation	Time Interval - Min.	Time From Launch - Min.
• INJECTION OF PRIMARY PAYLOAD INTO ORBIT		10.8
• ROLL, PITCH, ETC. MANEUVERS	73.3	
• BEGIN TRANSPOSITION AND DOCKING MANEUVERS		84.1
INITIATE SEPARATION DEPLOY ADAPTER TRANSLATE CSM FORWARD PITCH CSM 180° TRANSLATE CSM BACK COMPLETE SOFT DOCKING	10	
• COMPLETE DOCKING MANEUVER		94.1
	3	
• COAST TO S-IVB JETTISON		97.1
	2	
• JETTISON S-IVB/IU/SLA		99.1
	1	
• COMPLETE DOCKING AND JETTISONING		100.1
	1	
• DEPLOYMENT OF EXPERIMENTAL PAYLOADS POSSIBLE		101.1

2.3.3.1 Deployment Modes During Primary Mission Launch

An analysis of a typical Saturn IB launch trajectory was conducted in order to determine the modes of experimental payload deployment that are feasible and their limits of application during launch into orbit (up to primary payload injection).

In Figure 2-2, the variation and sensitivity characteristics of some primary orbital elements along a typical low-altitude, low-inclination orbital launch trajectory are shown as injection of the primary payload into its orbit is approached. An experimental payload physically but nonpropulsively separated from the vehicle would attain the given orbital elements. In this example, the time during which an experimental payload could be ejected and attain an individual orbit is limited to approximately two seconds prior to injection of the primary payload. The extreme sensitivity of perigee altitude to time before injection indicates that this mode of deployment (ejection without propulsion) is probably not desirable during the launch phase for this type of mission. Factors other than trajectory parameters further limit ejection during launch of the Saturn/Apollo configuration, e.g., physical separation from experimental payload locations defined in this study are not accessible for ejection until after the separation of vehicle/payload components.

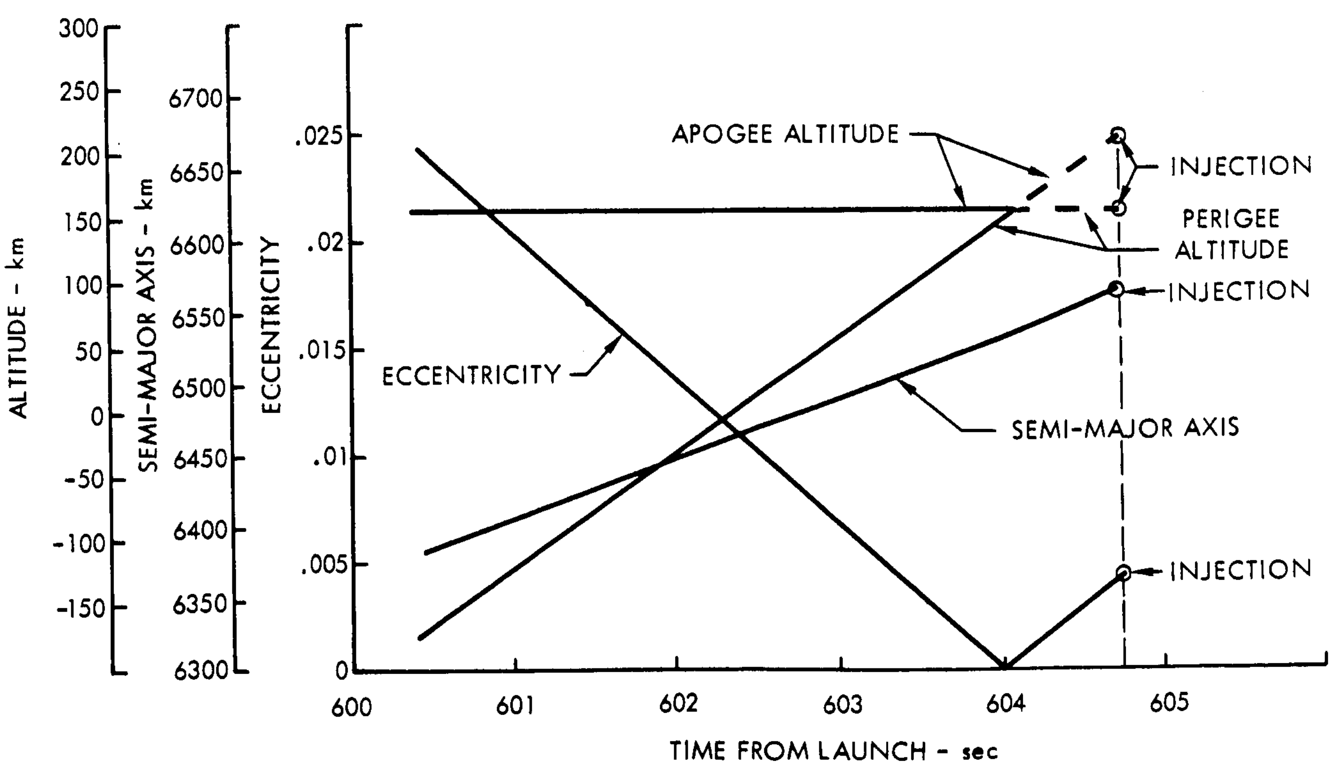


Figure 2-2 VARIATION OF ORBITAL ELEMENTS ALONG SATURN IB TYPICAL LAUNCH TRAJECTORY

2.3.3.2 Deployment Modes During Primary Mission Orbital Coast

During the primary mission orbital coast phase, experimental payloads may be deployed in various modes in order to achieve the maximum data acquisition objectives. For many experiments, the orbit achieved by the primary payload may be adequate for attainment of the majority of data acquisition objectives of the experiments, i.e., near maximum experiment effectiveness can be achieved in the orbit of the primary payload. Two basic modes of deployment are optional in this case, depending upon the physical environment required for the experiment: (1) the experimental payload remains fixed to the vehicle, in the location where it was placed prior to launch, or (2) the experimental payload is physically separated from the vehicle by some mechanism, e.g., a spring, which does not appreciably affect the orbital elements at the time of deployment. (Subsequent to deployment, the orbital elements of the vehicle and ejected experimental payload may differ because of perturbative forces such as atmospheric drag.)

For some experiments, the orbit achieved by the primary payload may be incompatible with the data acquisition objectives of the experiment. The logical mode of deployment in this case would be one in which propulsion is applied to the experimental payload in order to attain a more compatible orbit. A limited investigation was conducted to illustrate the effects of applying small impulsive velocity increments (ΔV's) to an experimental payload at deployment.

2.3.3.2.1 Effect of Example In-plane Deployment ΔV On Apogee and Perigee Altitudes (at Injection of Primary Payload). The results of an investigation to determine the effects of propulsive deployment on apogee and perigee altitudes are shown in Figure 2-3. Impulsive velocity increments (ΔV's) were assumed to be applied normal to the injection velocity vector, in the plane of the orbit. The time of ΔV application was assumed to be at the instant of injection of the primary payload into its orbit. Injection conditions are those of the typical launch trajectory given in Figure 2-1.

It is noted that in this example ΔV application, an appreciable decrease in perigee altitude and increase in apogee altitude can be achieved with small ΔV's (up to 100 m/sec).

2.3.3.2.2 Effect of Example In-plane and Out-of-plane Deployment ΔV on Orbital Elements (after Injection of Primary Payload). The results of the previous example deployment investigations have shown that deployment of an experimental payload before or at ejection of the primary payload has limited application or is physically impossible. In particular, in the case of the payload cavities that have been defined for the example configuration for the study (Saturn IB/Apollo), deployment modes that require

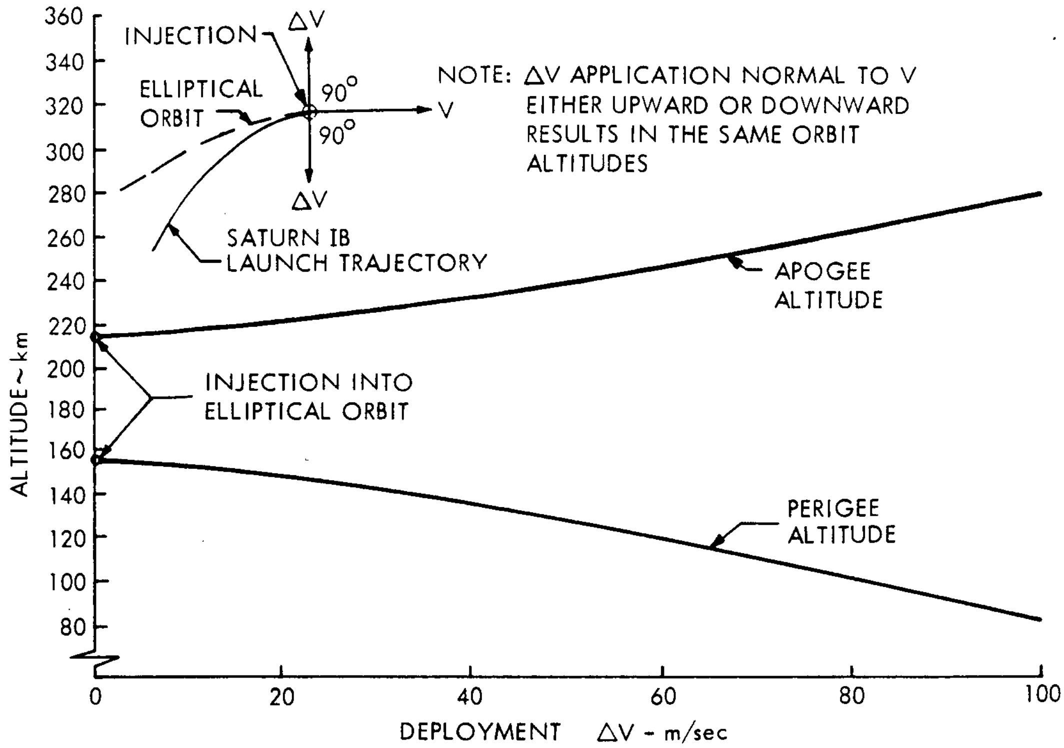


Figure 2-3 EFFECT OF EXAMPLE ΔV APPLICATION ON APOGEE AND PERIGEE ALTITUDES AT INJECTION OF PRIMARY PAYLOAD

physical separation of the experimental payload from the vehicle must occur after injection and actual separation of the LEM/CSM from the SIVB LEM adapter.

In Figure 2-4, data are given to illustrate the major effects of applying small ΔV's in each of the in-plane and out-of-plane orthogonal directions (i.e., tangential, normal, and lateral). These data are given for the reference elliptical orbit of the primary payload. Since the initial orbit is elliptical, the effects of ΔV application vary as a function of time after launch (position in the orbit).

As illustrated in the upper right diagram in Figure 2-4, the tangential direction lies along the velocity vector, the normal direction is perpendicular to the tangential direction and in the plane of the orbit, and the lateral direction completes the right-handed cartesian coordinate system.

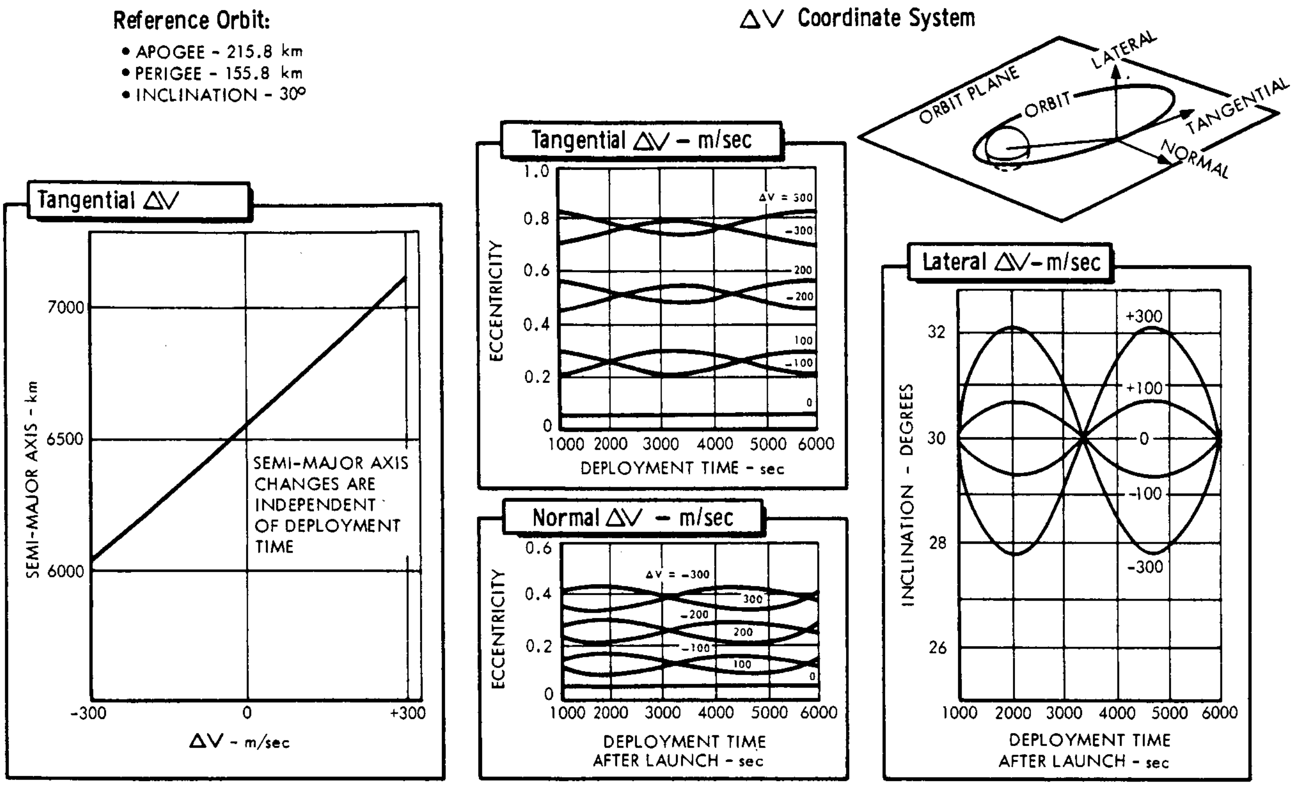


Figure 2-4 EFFECTS OF DEPLOYMENT ΔV COMPONENTS ON ORBITAL ELEMENTS

The major effects of applying small impulsive velocity increments (≤300 m/sec) are summarized as follows:

1. Tangential ΔV affects the value of the semi-major axis and the eccentricity of the orbit. Note that semi-major axis changes are independent of the point of ΔV application in the orbit (i.e., time of deployment). However, eccentricity is a function of the time of deployment (for an elliptical orbit).

2. Normal ΔV affects the value of orbit eccentricity. The effect is a function of deployment time.

3. Lateral ΔV affects the orbit inclination and the longitude of the ascending node. The effects are dependent on deployment time. Maximum change of inclination occurs if the ΔV is applied at the nodes, and maximum nodal shift occurs at the maximum latitude point. No inclination change occurs if it is applied at the position of maximum latitude, and no nodal shift occurs at the nodal point. As the point of ΔV application is moved from the node toward the maximum latitude point, there is less change in the inclination and more change in the longitude of the ascending node.

2.4 REFERENCES

2.1 Saturn Mission Plan and Technical Information Checklist, NASA/MSFC, July 1965. (U)

2.2 Cremin, J. W. and Gillis, W. M., SA-201 Launch Vehicle Reference Trajectory, NASA TM X-53242, 15 April 1965. (U)

2.3 Nominal Reference Trajectory for the Saturn IB-SA-203 with Revised Control Weights, Chrysler Technical Bulletin TB-AE-65-128, Space Division, Chrysler Corporation, 1 February 1965. (C)

2.4 Saturn IB SA-204 and Subsequent Design Trajectories, Chrysler Technical Bulletin TB-AE-65-117, February 1965. (C)

2.5 Reference Trajectory for the Saturn IB-SA-206, Chrysler Technical Bulletin, TB-AE-65-132, Space Division, Chrysler Corporation, 9 February 1965. (C)

2.6 Dispersion Analysis, Nominal, and Design Trajectories for the Saturn IB Operational Vehicle (SA-207) with Uprated H-1 Engines, NASA/MSFC Memorandum R-AERO-DAP-59-65, 30 June 1965. (C)

SECTION 3

EXPERIMENT CAVITY

DESCRIPTION METHODOLOGY

3.1 GENERAL

The baseline configuration selected for use in this study is the Saturn IB/Apollo with Command Module, Service Module, and Lunar Excursion Module. Whenever information was available, the SA-207 configuration was used in describing potential experiment locations. The following general areas of this vehicle were investigated for use as potential experiment locations:

1. LEM Adapter Fairing (Spacecraft LEM Adapter)

2. Instrument Unit

3. S-IVB Stage forward skirt

4. S-IVB Stage LH_2 tank

5. S-IVB Stage aft skirt

6. External pods.

Items 4 and 5 were eliminated from further consideration because of their extreme environments (temperature, vibration, etc.) and their limited mass and volume capabilities; item 6 was not considered further because it was not compatible with the scope of the study. The areas chosen for consideration in this study (LEM Adapter Fairing, Instrument Unit, and S-IVB Stage forward skirt) contain nearly all of the available volume in that portion of the Saturn IB/Apollo vehicle that is injected into Earth orbit. Furthermore, these areas provide the capabilities (load carrying, deployment, accessibility, etc.) that are required for the successful accomplishment of the majority of the in-flight experiments.

In order to simplify the identification of the potential experiment locations, the general areas under consideration were subdivided into seven zones as shown in Figure 3-1. Each zone contains several individual cavities which are potential locations for in-flight experiments. A total of 53 cavities were defined, and each of these cavities was described in terms of the following:

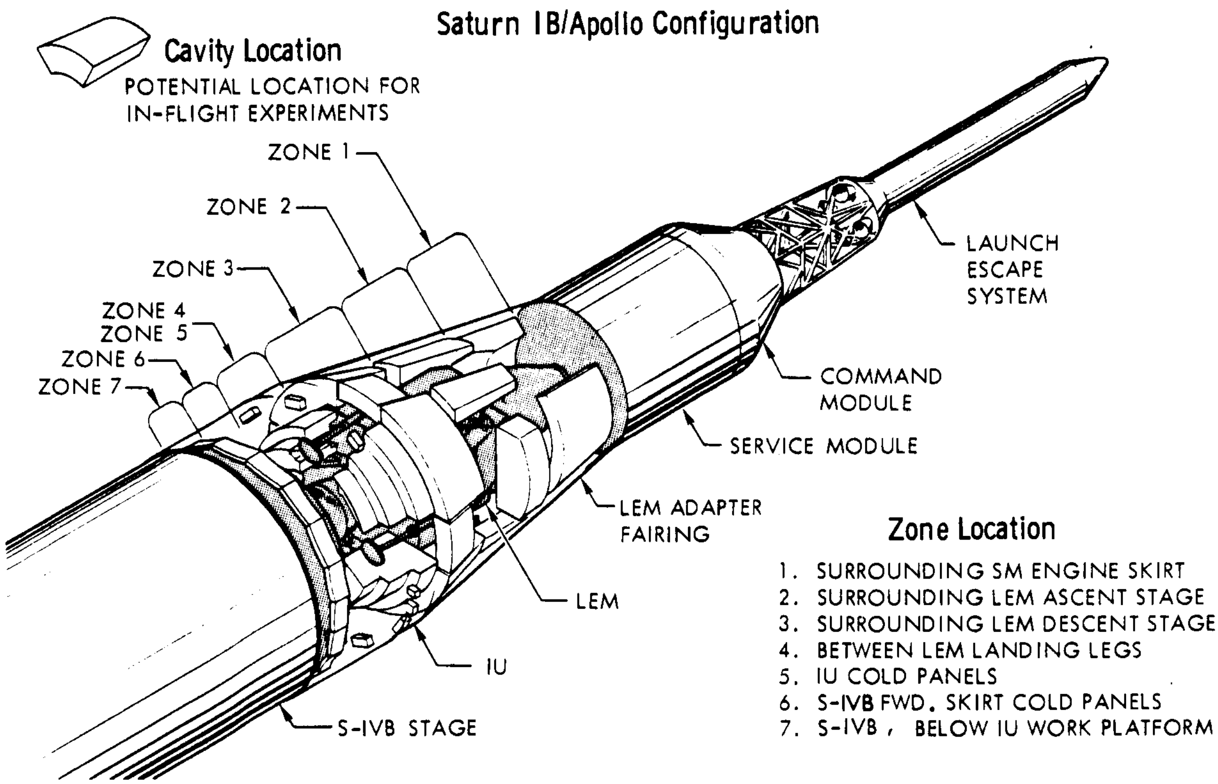


Figure 3-1 IN-FLIGHT EXPERIMENT CAVITY LOCATIONS

1. Volume/Geometry
2. Mass capacity
3. Deployment capabilitiy
4. Environment.

3.2 SELECTION AND IDENTIFICATION OF ZONES

The zone divisions which are shown in Figure 3-1 were apportioned so that the cavities contained in each zone would be similar in terms of location, accessibility, installation requirements, deployment capability, and environment. The zones are numbered in sequence, beginning with the zone nearest the Service Module and progressing aft to the S-IVB stage. Separate zones are provided for the Instrument Unit and S-IVB Stage cold panels.

The locations of the seven zones within the Saturn vehicle are described briefly as follows:

1. Zone 1 is located within the LEM Adapter Fairing (Spacecraft LEM Adapter) and surrounds the Service Module engine nozzle. It extends from the work platform around the upper portion of the LEM Ascent Stage, Saturn IB/Apollo Station 1888, to the intersection of the LEM Adapter Fairing and the Service Module, Station 2035.

2. Zone 2 is located within the LEM Adapter Fairing and surrounds the LEM Ascent Stage. It extends from the work platform between the LEM Ascent and Descent Stages, Station 1794, to the lower face of Zone 1, Station 1888.

3. Zone 3 is located within the LEM Adapter Fairing and surrounds the LEM Descent Stage. It extends from the lower surface of the LEM Descent Stage primary structure, Station 1722, to the lower face of Zone 2, Station 1794.

4. Zone 4 is located within the LEM Adapter Fairing, the IU, and the S-IVB forward skirt. It includes the area between the LEM landing legs and around a portion of the LH_2 tank forward dome. Zone 4 extends from the lower surface of the S-IVB work platform, Station 1633, to the lower face of Zone 3, Station 1722.

5. Zone 5 is located on the Instrument Unit cold panels.

6. Zone 6 is located on the S-IVB forward skirt cold panels.

7. Zone 7 is located within the S-IVB forward skirt below the work platform. It extends from the lower edge of the skirt, Station 1541, to the lower face of Zone 4, Station 1633.

3.3 SELECTION AND IDENTIFICATION OF CAVITIES

As previously mentioned, each of the seven zones of the vehicle contains several individual cavities which are potential locations for in-flight experiments. These cavity shapes were obtained by providing the following clearances from the Saturn IB/Apollo vehicle:

1. A 6-inch clearance was allowed between each cavity and the LEM, Service Module, S-IVB Tank, and all Work Platforms to provide adequate space for installation and maintenance of the experiment package.

2. A 3-inch clearance was provided between each cavity and the LEM Adapter Fairing to provide for experiment mounting structure.

3. Adequate clearance was provided for extraction of the LEM from the Adapter Fairing during an orbital mission.

4. Direct attachment to the cold panels in the Instrument Unit and the S-IVB Stage forward skirt was assumed.

5. Clearances from existing components on the cold panels were per NASA report (Reference 3-1).

It is recognized that the cavities in Zones 1, 2, and 3 may interfere with the work space provided above the work platforms in the LEM Adapter Fairing. However, it is felt that, by the use of proper sequencing of the experiment installation, much of this space can be made available for in-flight experiments. The work envelope for the S-IVB Work Platform and the related Component Handling Equipment is not obstructed by any of the cavities.

As shown in Figure 3-2, the cavities in each zone are identified as a separate dash number of that zone. Cavities are numbered clockwise looking forward on the vehicle with the numbers beginning at position 1 which is the down position in Earth orbit. A drawing has been prepared (Fig. 3-3) on which each of the 53 cavities defined in this study is identified and located.

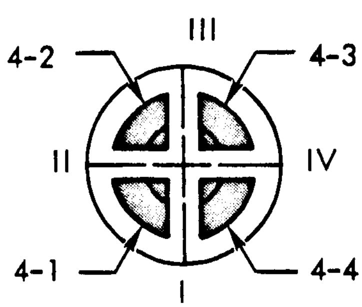
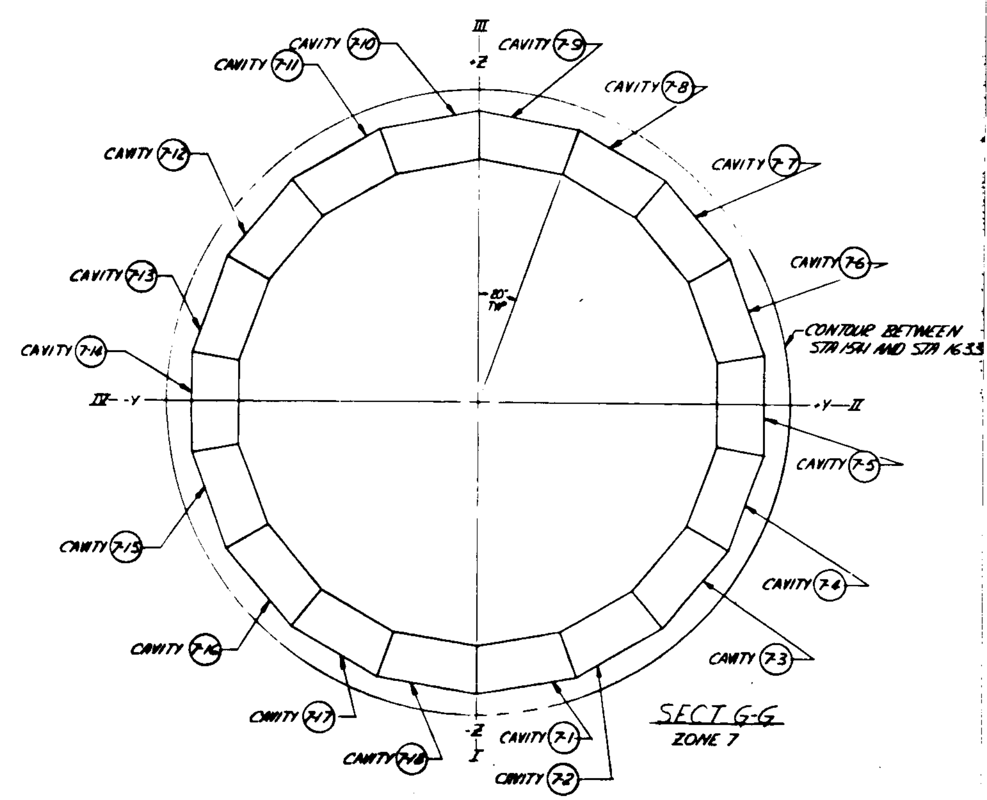


Figure 3-2 ZONE 4 CAVITIES

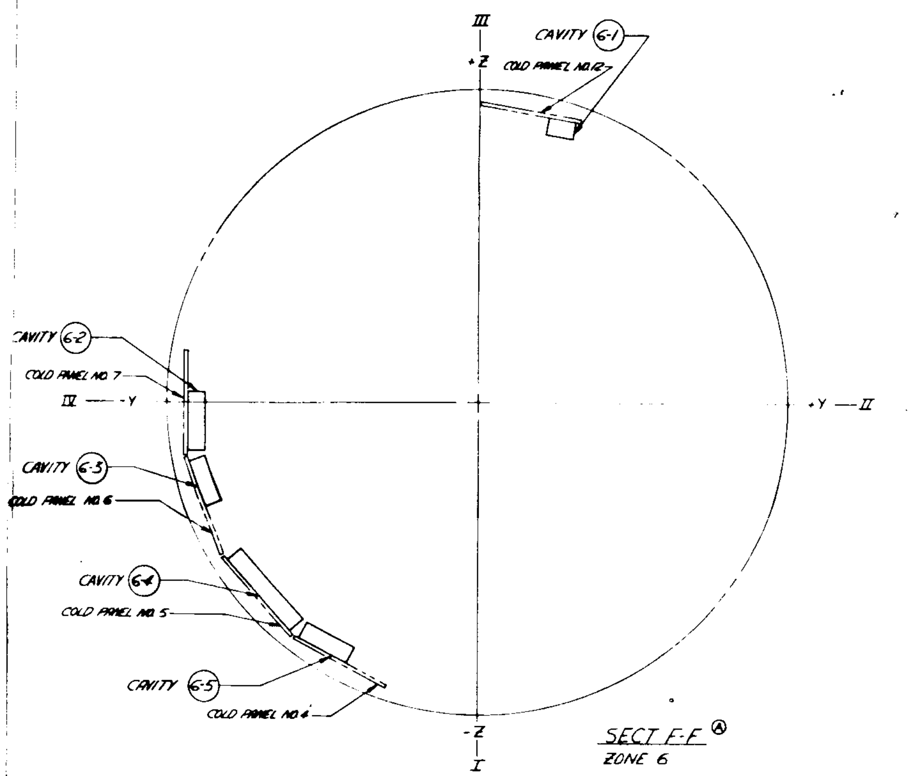
An isometric drawing, including dimensions and orientation, was made for each of the 53 cavities. These drawings are contained in Appendix A of this volume. The V, R, and L axes system is used in defining the dimensions and orientations of the cavities. The V axis is generally parallel to the launch vehicle longitudinal axis, the R axis is generally normal to the external contour of the vehicle, and the L axis is 90 degrees to both the V and R axes.

3.4 VOLUME/GEOMETRY

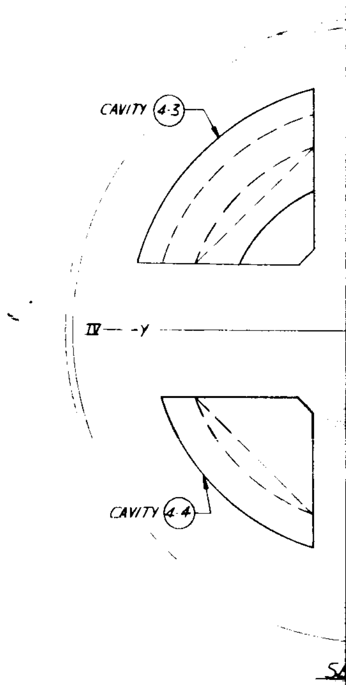
The geometry of each cavity was described in terms of its capacity to contain three standard geometrical shapes, the rectangular parallelepiped, the sphere, and the cyclinder. This approach is compatible with the methods described in Section 4 for representing experiment geometry

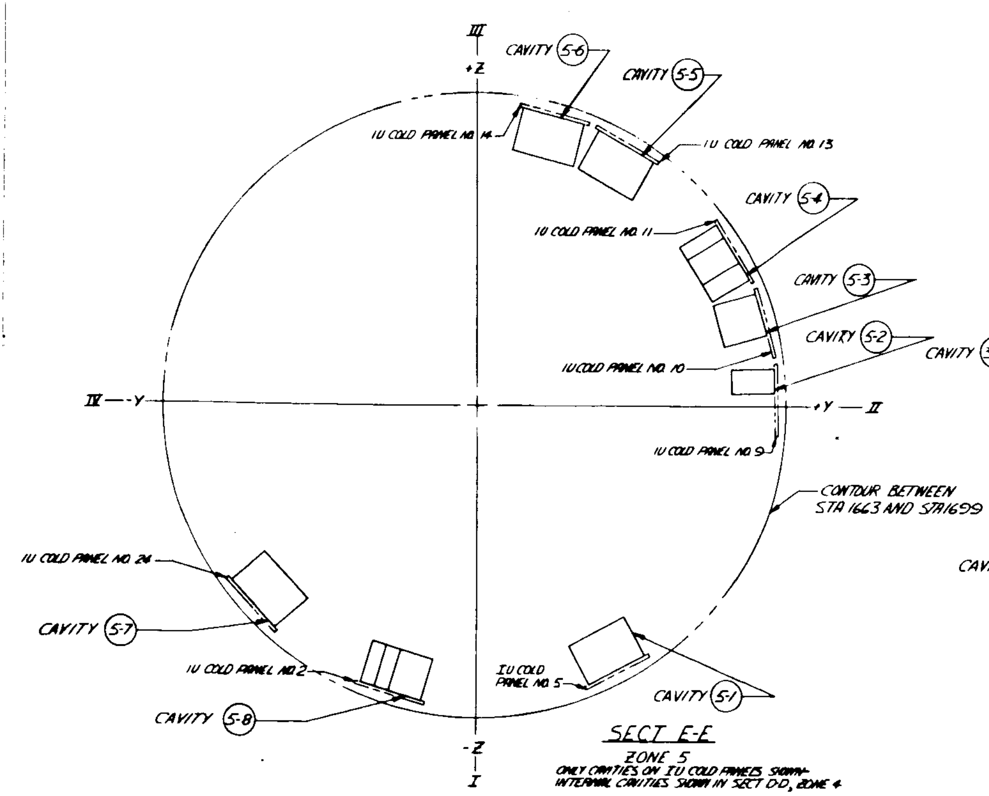


SECT G-G
ZONE 7

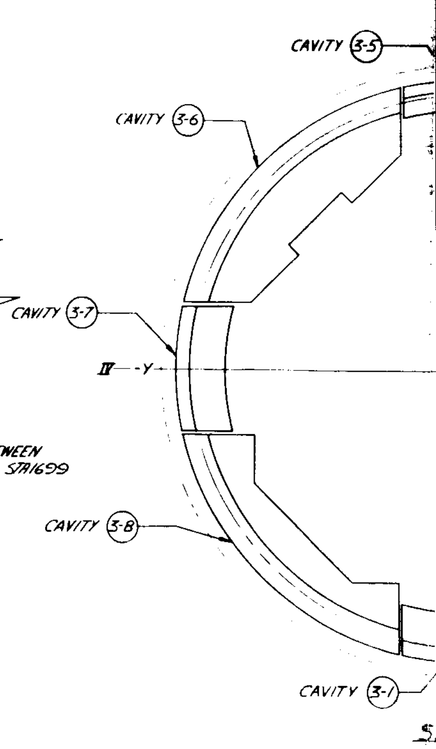


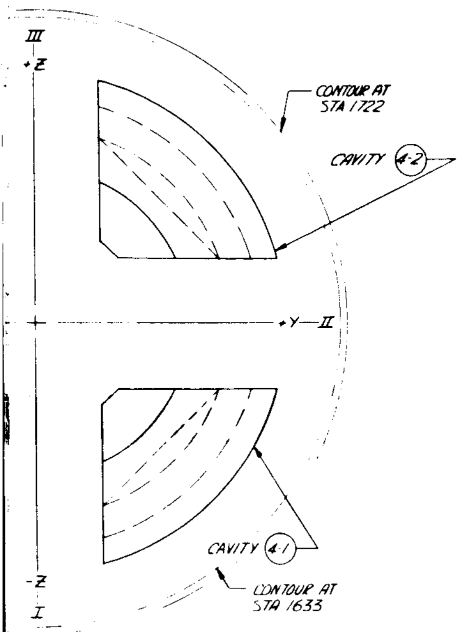
SECT F-F Ⓐ
ZONE 6



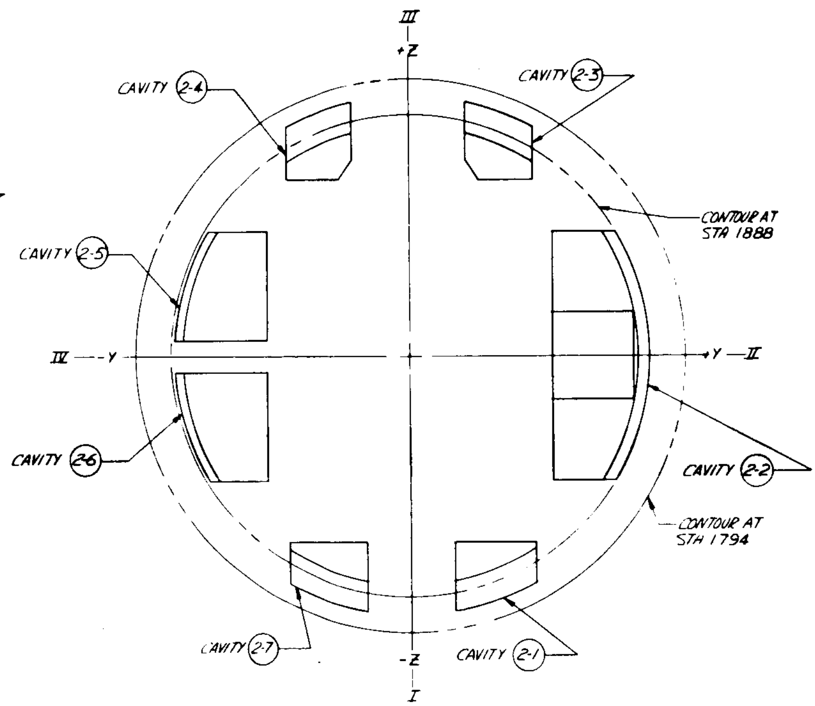


SECT E-E
ZONE 5
ONLY CAVITIES ON IU COLD PANELS SHOWN-
INTERNAL CAVITIES SHOWN IN SECT D-D, ZONE 4

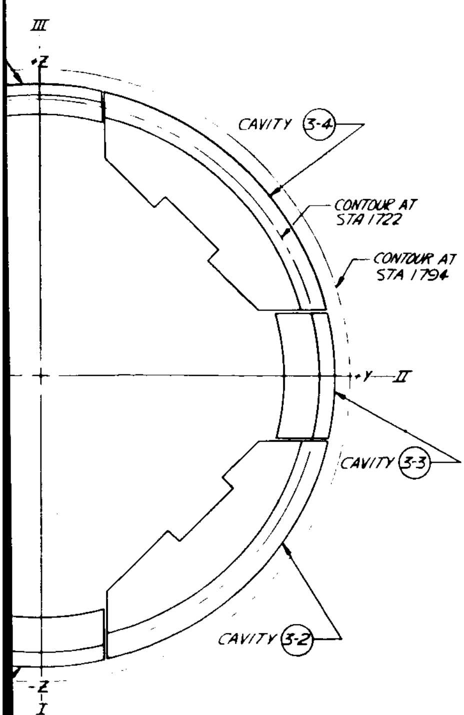




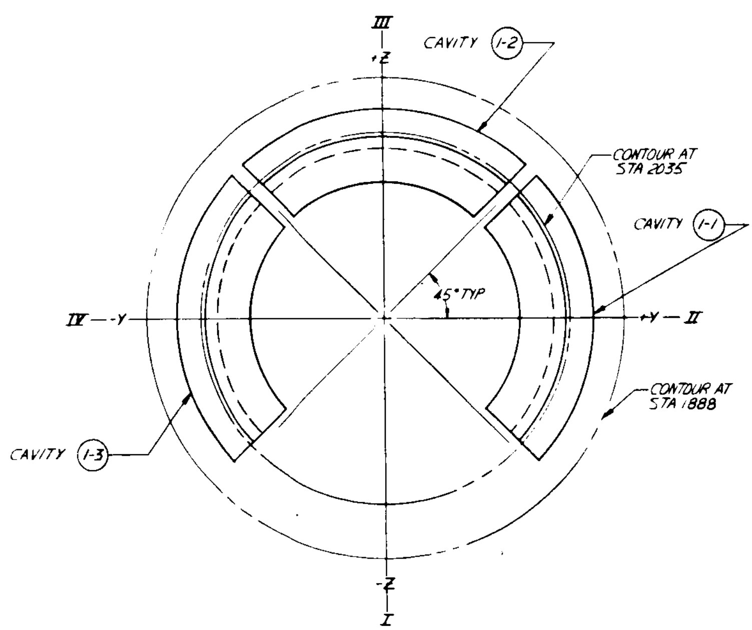
ECT D-D
ZONE 4
S AVAIL ON IU COLD PANELS
IVB COLD PANELS SHOWN
T E-E AND SECT F-F



SECT B-B
ZONE 2

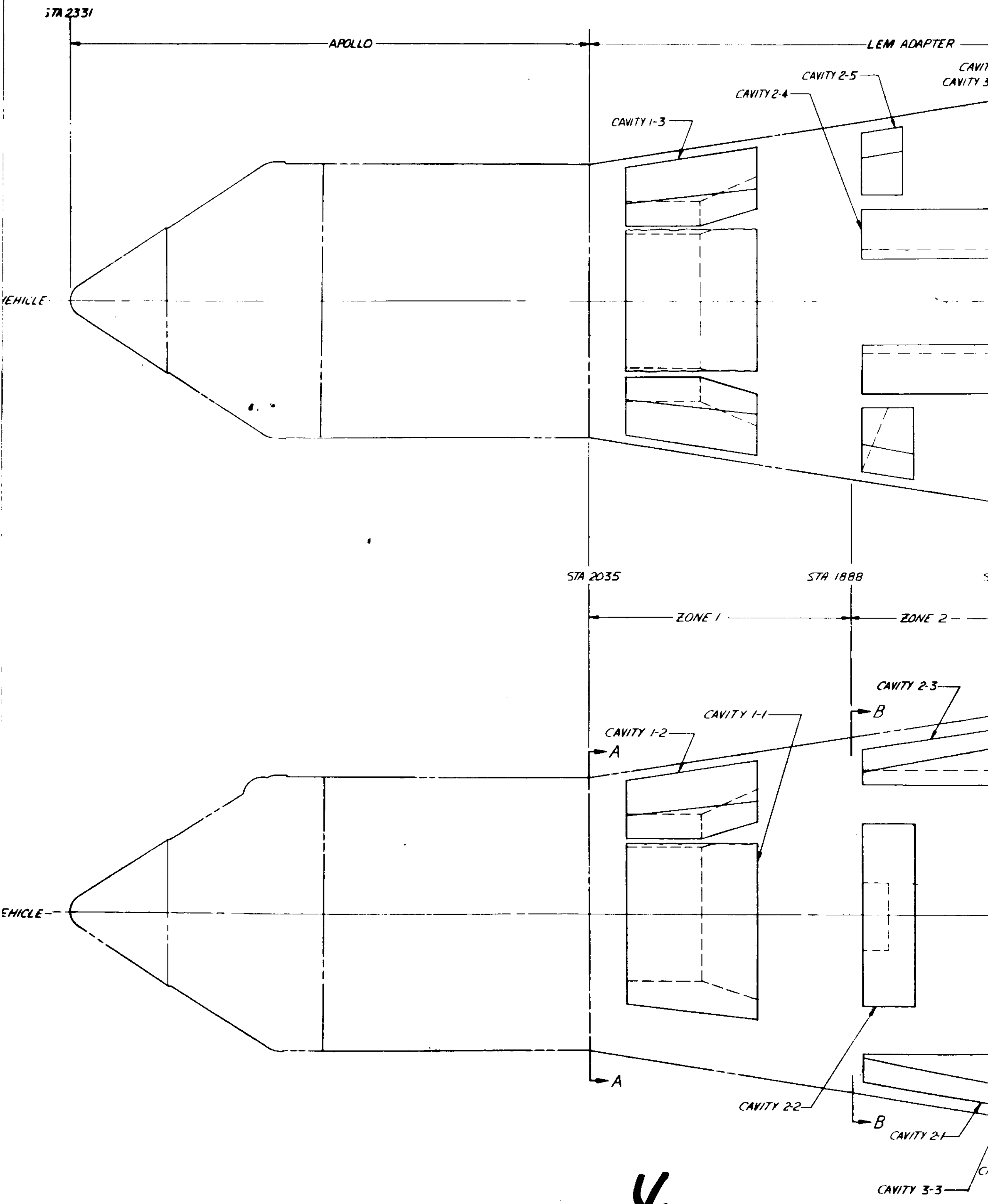


ECT C-C
ZONE 3



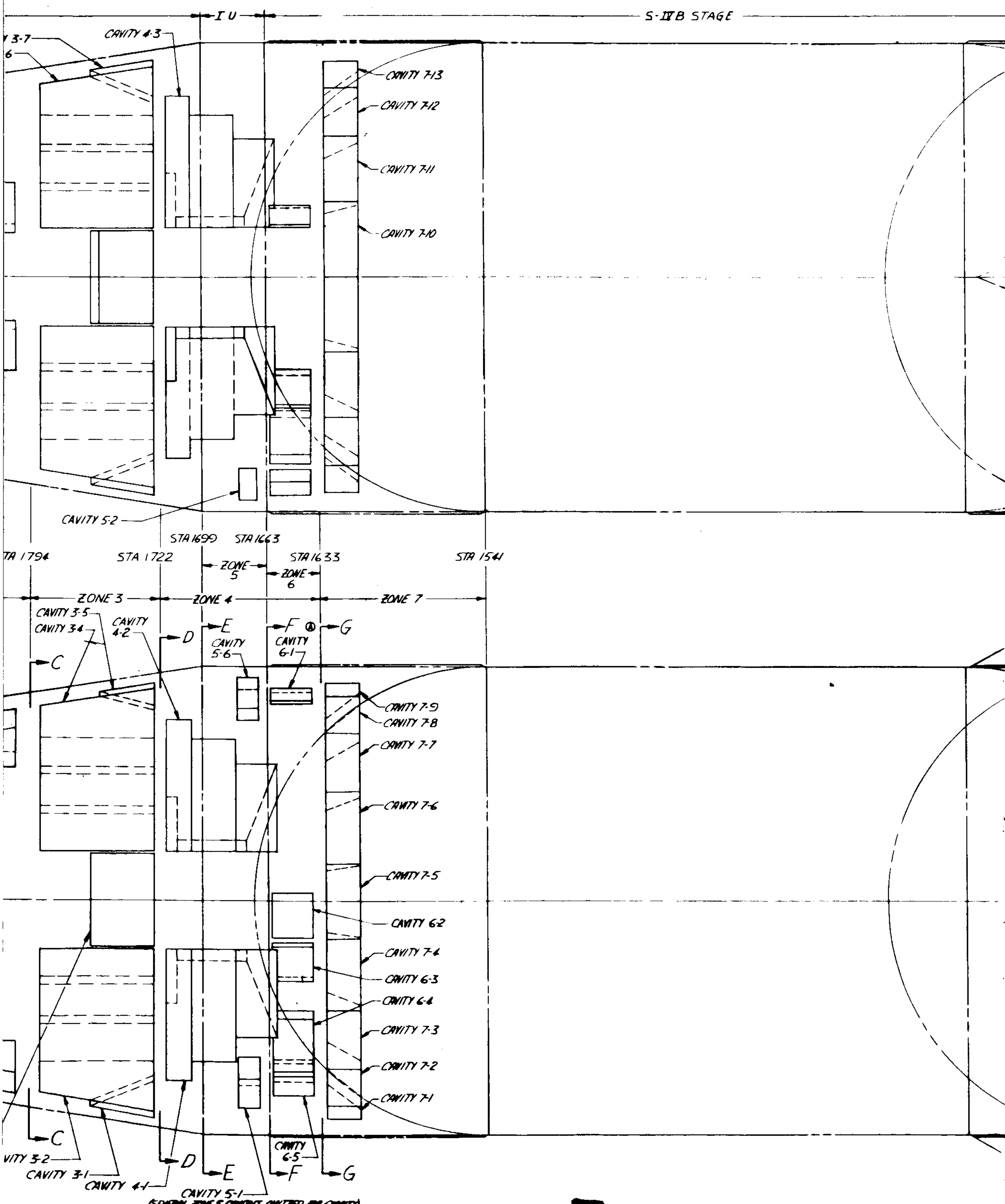
SECT A-A
ZONE 1

3

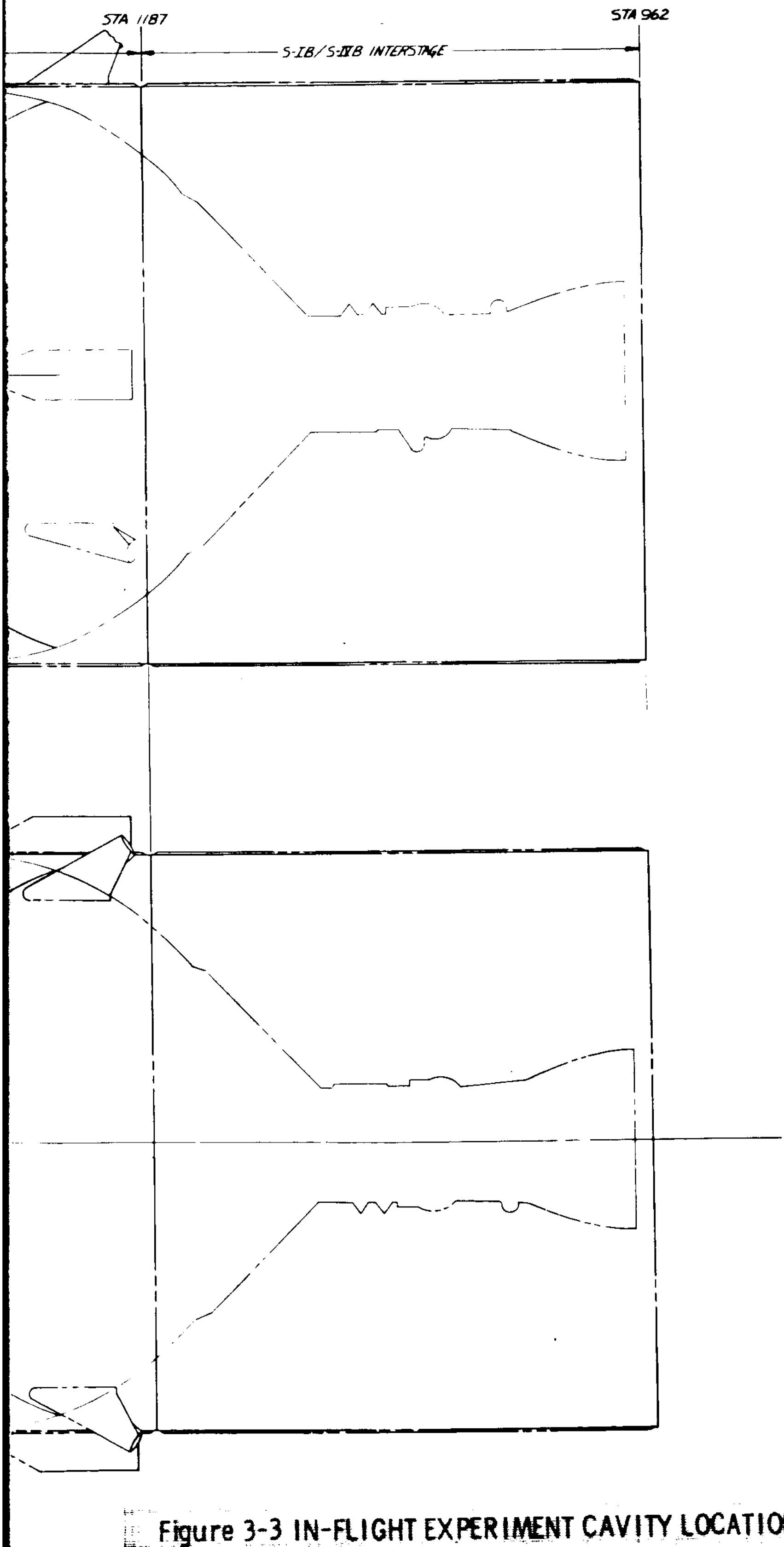


STA 2331
APOLLO
LEM ADAPTER
CAVITY 2-5
CAVITY 2-4
CAVITY 1-3
VEHICLE
STA 2035
STA 1888
ZONE 1
ZONE 2
CAVITY 2-3
CAVITY 1-1
CAVITY 1-2
A
B
VEHICLE
CAVITY 2-2
CAVITY 2-1
CAVITY 3-3

4


I U
S-IVB STAGE
CAVITY 4-3
CAVITY 7-13
CAVITY 7-12
CAVITY 7-11
CAVITY 7-10
CAVITY 5-2
STA 1699
STA 1663
STA 1722
STA 1633
STA 1541
ZONE 5
ZONE 6
ZONE 3
ZONE 4
ZONE 7
CAVITY 3-5
CAVITY 3-4
CAVITY 4-2
CAVITY 5-6
CAVITY 6-1
CAVITY 7-9
CAVITY 7-8
CAVITY 7-7
CAVITY 7-6
CAVITY 7-5
CAVITY 6-2
CAVITY 7-4
CAVITY 6-3
CAVITY 6-4
CAVITY 7-3
CAVITY 7-2
CAVITY 7-1
CAVITY 6-5
CAVITY 3-1
CAVITY 4-1
CAVITY 5-1
(SEVERAL ZONE 5 CAVITIES OMITTED FOR CLARITY)

5



"A" CHG - ADDED ZONE 6 CAVITIES TO SECT F-F AND SIDE VIEWS. 12-2-65

6

Figure 3-3 IN-FLIGHT EXPERIMENT CAVITY LOCATIONS FOR SATURN IB/APOLLO by the same standard shapes. Definition of the standard shapes capacity of the cavities was complicated by the fact that most of the cavities contain tapers and contours. As shown in Figure 3-4, the capacities were defined in the following manner:

EACH CAVITY DEFINED BY CAPACITY TO CONTAIN CERTAIN STANDARD GEOMETRICAL SHAPES

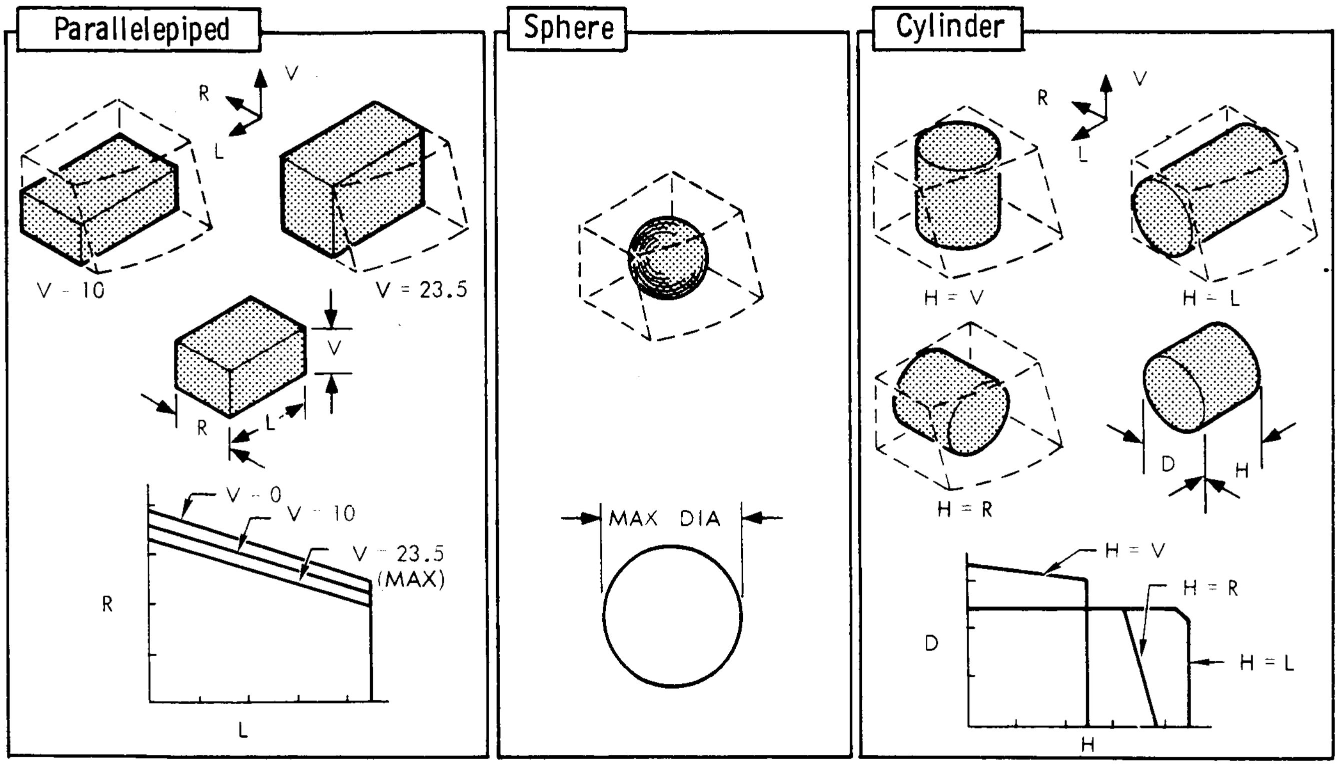


Figure 3-4 CAVITY GEOMETRY

1. The capacity to contain parallelepipeds was described by curves in which the R dimension is a function of the L dimension for various values of V. The R, L, and V dimensions are the dimensions along the R, L, and V axes previously described. In Figure 3-4, several parallelepipeds are shown contained within a typical cavity, and a curve is presented to depict all possible variations of the R, L, and V dimensions which can be contained within that cavity.

2. The capacity to contain spheres was described by the maximum diameter sphere that can be contained within a cavity.

3. The capacity to contain cylinders was described by curves for which the diameter D is a function of the length H for each of three orientations: H parallel to the R axis, H parallel to the L axis, and H parallel to the V axis. In Figure 3-4, a cylinder is shown in a typical cavity in the three orientations; the curve presented reflects possible variations of H and D for the three orientations.

A summary of the volume and geometry (standard shapes capacity) of the 53 cavities is presented in Figure 3-5. The volumes range from 163,966 cubic inches for cavities 1-1, 1-2, and 1-3 to 1800 cubic inches for cavity 5-2.

3.5 MASS CAPACITY

The mass capacities shown in Figure 3-5 for cavities in Zones 4, 6, and 7 are based on values obtained from the Saturn IB Payload Planner's Guide (Reference 3-2). The values of 2500 and 1000 pounds for Zones 4 and 7 respectively are predicated on a Mode I operation in which only one experiment is located on the vehicle. These values represent the total load-carrying capability of that particular vehicle zone. For a Mode II operation in which multiple experiments are located on the vehicle, the capability would be equal to the capability of the zone less the mass of the experiments already located in that particular zone. The mass capacities of the cavities in Zone 6 are based on the load-carrying capability of the individual cold panels and are applicable to both Mode I and Mode II operations.

The mass capacities for Zone 5 were obtained from MSFC document Preliminary Definition of Saturn Instrument Unit and S-IVB Support Capability for Extended Apollo Earth-Orbit Experiments (Reference 3-1). These capacities were also determined by the load-carrying capability of the individual cold panels and are applicable to both Mode I and Mode II.

In order to determine the mass capacity of cavities in Zones 1, 2, and 3, an investigation was made to determine the critical design conditions and the existing margins of safety for the LEM Adapter Fairing (Spacecraft LEM Adapter). The majority of the data used in the investigation were obtained from Apollo Spacecraft Structural Analysis of the SLA (Reference 3-3).

The critical design conditions of the Fairing structure are

1. Maximum $q\alpha$ - the point of maximum body loads. It occurs at room temperature.

2. End of first stage boost - the time of maximum temperature

3. First stage separation - body tension loads produced by this condition. It occurs at essentially the end of boost temperature.

It was assumed that Saturn IB loading is approximately 65 to 85 percent of the Saturn V loading, and that the Block II LEM Adapter

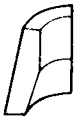
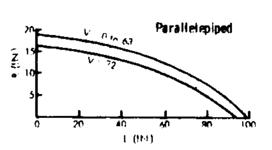
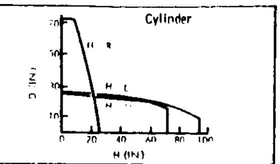
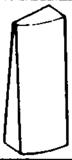
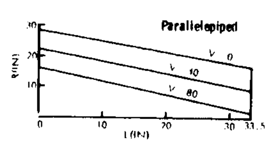
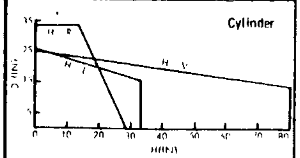
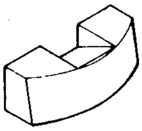
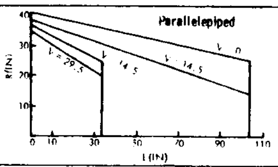
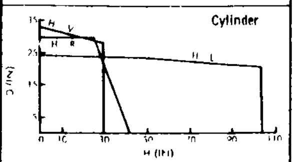
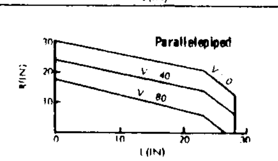
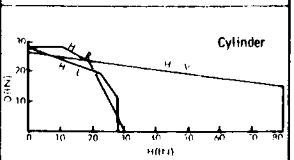
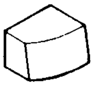
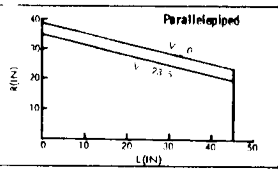
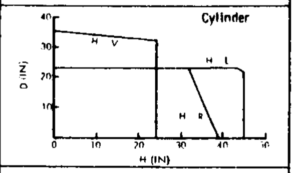
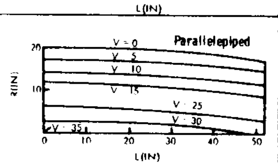
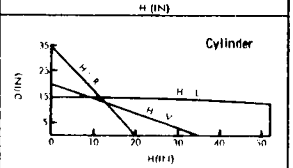
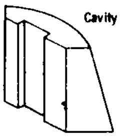
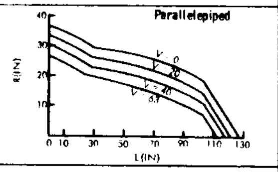
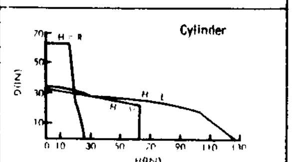
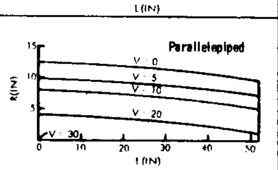
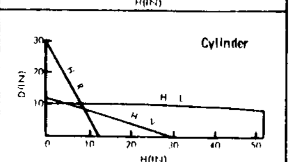
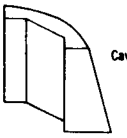
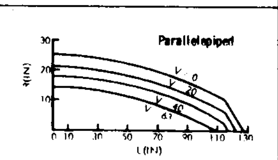
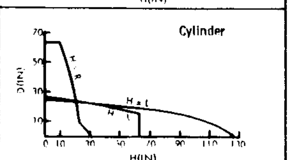
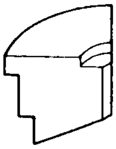
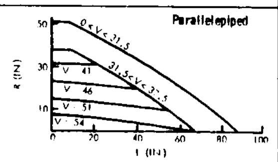
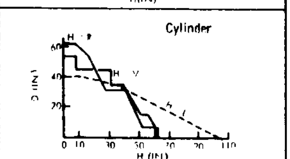
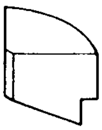
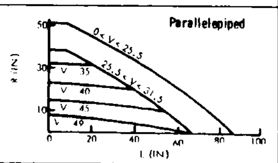
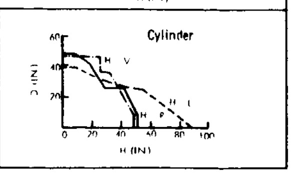
ZONE 1	 Cavity 1-1, 1-2, & 1-3	• Volume ..... 163,966 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 24.5 in		
ZONE 2	 Cavity 2-1 & 2-7	• Volume ..... 42,560 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 21.0 in		
ZONE 2	 Cavity 2-2	• Volume ..... 92,030 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 29.5 in		
ZONE 2	 Cavity 2-3 & 2-4	• Volume ..... 43,300 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 25.0 in		
ZONE 2	 Cavity 2-5 & 2-6	• Volume ..... 30,000 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 23.5 in		
ZONE 3	 Cavity 3-1, 3-3, 3-7	• Volume ..... 18,200 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 15.0 in		
ZONE 3	 Cavity 3-2, 3-4, & 3-6	• Volume ..... 171,039 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 28.0 in		
ZONE 3	 Cavity 3-5	• Volume ..... 9,350 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 10.0 in		
ZONE 3	 Cavity 3-8	• Volume ..... 132,500 in^3 • Mass Capacity ..... 1,000 lbs • Sphere Diameter ..... 22.5 in		
ZONE 4	 Cavity 4-1, 4-2, & 4-3	• Volume ..... 138,080 in^3 • Mass Capacity ..... 2,500 lbs • Sphere Diameter ..... 38.5 in		
ZONE 4	 Cavity 4-4	• Volume ..... 94,280 in^3 • Mass Capacity ..... 2,500 lbs • Sphere Diameter ..... 38.5 in		

Figure 3-5 CAVITY VOLUME, GEOMETRY AND MASS CAPACITY SUMMARY

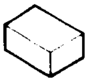
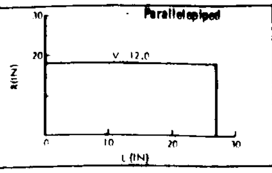
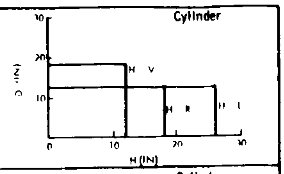
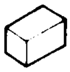
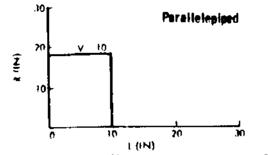
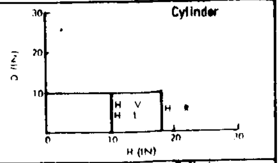
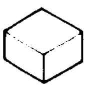
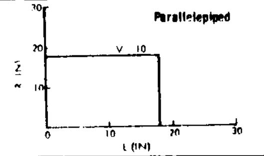
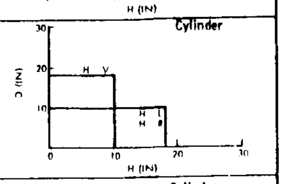
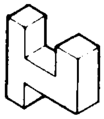
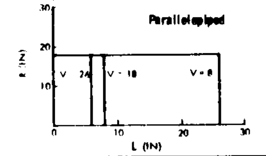
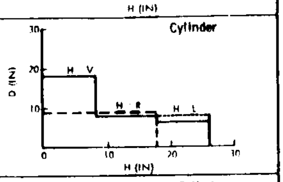
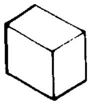
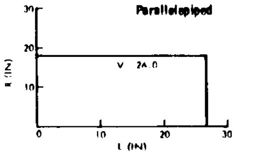
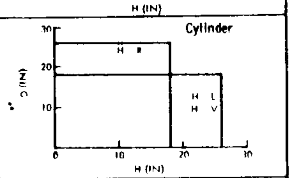
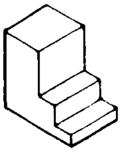
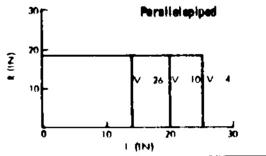
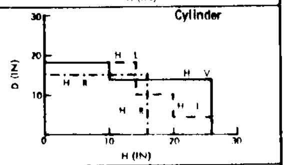
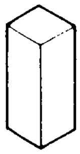
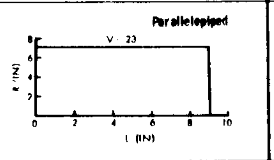
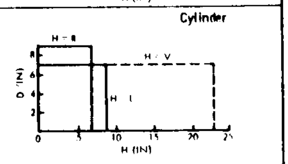
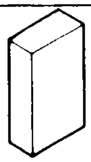
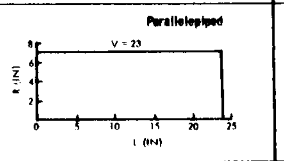
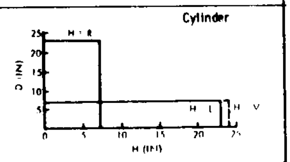
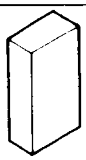
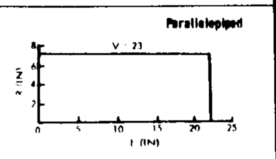
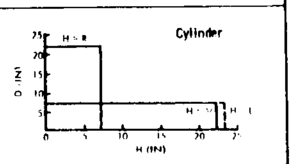
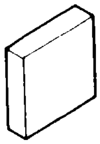
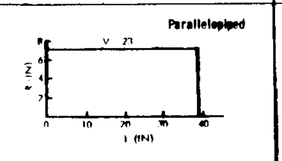
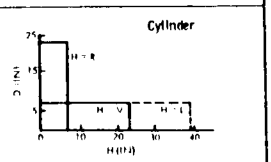
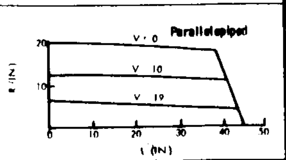
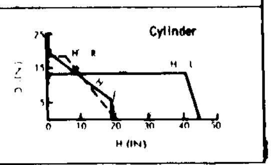
Zone	Cavity	Data	Parallelepiped	Cylinder
ZONE 5	 Cavity 5-1 & 5-6	• Volume ... 5,616 in^3 • 5-1 Mass Capacity ... 150 lbs • 5-6 Mass Capacity ... 297 lbs • Sphere Diameter ... 12.0 in		
ZONE 5	 Cavity 5-2	• Volume ... 1,800 in^3 • Mass Capacity ... 276 lbs • Sphere Diameter ... 10.0 in		
ZONE 5	 Cavity 5-3	• Volume ... 3,240 in^3 • Mass Capacity ... 280 lbs • Sphere Diameter ... 10.0 in		
ZONE 5	 Cavity 5-4	• Volume ... 6,950 in^3 • Mass Capacity ... 208 lbs • Sphere Diameter ... 9.3 in		
ZONE 5	 Cavity 5-5 & 5-7	• Volume ... 12,186 in^3 • Mass Capacity ... 330 lbs • Sphere Diameter ... 18.0 in		
ZONE 5	 Cavity 5-8	• Volume ... 8,064 in^3 • Mass Capacity ... 286 lbs • Sphere Diameter ... 14.5 in		
ZONE 6	 Cavity 6-1	• Volume 1,449 in^3 • Mass Capacity 50 lbs • Sphere Diameter 7.0 in		
ZONE 6	 Cavity 6-2	• Volume 3,864 in^3 • Mass Capacity 130 lbs • Sphere Diameter 7.0 in		
ZONE 6	 Cavity 6-3 & 6-5	• Volume 3,542 in^3 • Mass Capacity 125 lbs • Sphere Diameter 7.0 in		
ZONE 6	 Cavity 6-4	• Volume 6,279 in^3 • Mass Capacity 150 lbs • Sphere Diameter 7.0 in		
ZONE 7	 Cavity 7-1 thru 7-18	• Volume ... 9,610 in^3 • Mass Capacity ... 1,000 lbs • Sphere Diameter ... 13.2 in		

Figure 3-5 CAVITY VOLUME, GEOMETRY AND MASS CAPACITY SUMMARY CONT'D

Fairings will be capable of Saturn V loading. Therefore, for the purposes of this study, it was estimated that approximately 1000 pounds of secondary payloads can be carried on each quarter segment of the fairing. This assumes that the method of mounting the experiment will provide for correct distribution of the loads into the fairing. Because the fairing is a redundant structure, a computer solution would be required to determine the actual loadings after installation of an experiment. The 1000-pound capacity represents the load-carrying capacity of each segment of the fairing and is predicated on a Mode I operation. For a Mode II operation the capacity of a cavity would be equal to 1000 pounds less the mass of the experiment already located on that particular adapter fairing segment.

3.6 DEPLOYMENT CAPABILITY

The "cavity deployment capability" refers to the ability of a cavity to contain experiments that require exposure to vacuum, extension of an experiment component from the launch vehicle, separation of the experimental payload from the launch vehicle, or separation of a data recovery capsule. This capability is limited by the launch vehicle configuration and the location of the cavity on the vehicle. The minimum deployment capability shown in Figure 3-6, occurs prior to

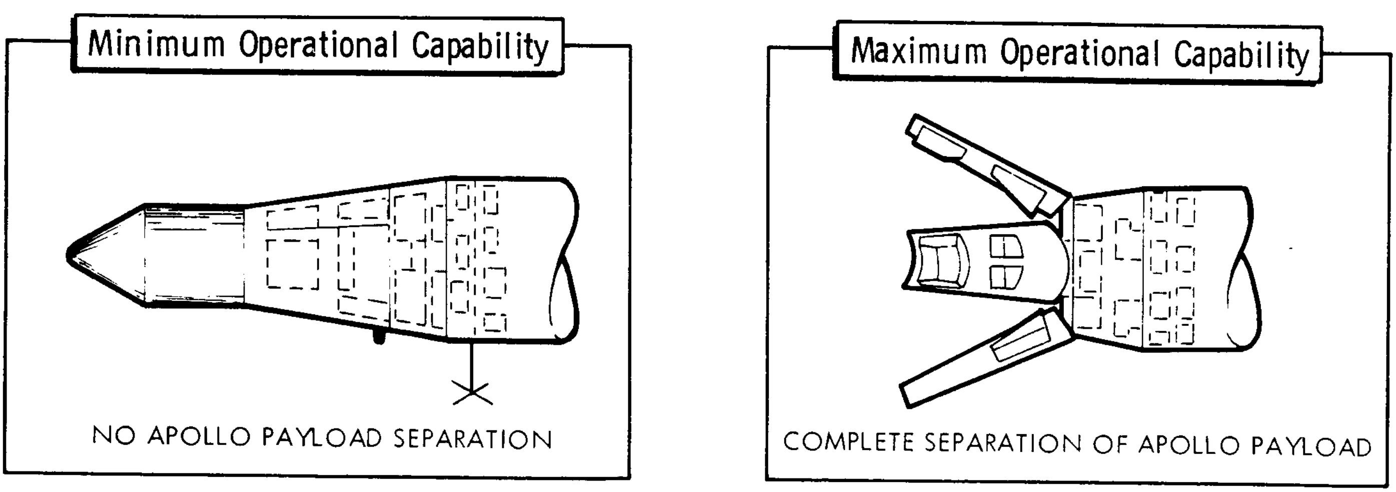


Figure 3-6 CAVITY DEPLOYMENT CAPABILITY

separation of the Apollo payload. The maximum deployment is obtained after complete separation of the Apollo payload. In Section 4 the experiment deployment requirements are described by the following six deployment modes:

Mode 0 - The experimental payload remains on the vehicle, requires extension of only an antenna, and does not require exposure to vacuum.

Mode 1 - The experiment remains on the vehicle, requires extension of only an antenna, and requires exposure to vacuum.

Mode 2 - The experiment remains on the vehicle and requires extension of components other than antennas.

Mode 3 - The experimental payload is ejected from the vehicle, and no payload propulsion is required.

Mode 4 - The experimental payload is ejected from the vehicle, and payload propulsion is required.

Mode 5 - The experiment remains on the launch vehicle, but ejection of one or more data recovery capsules is required.

The deployment capability of the cavities can be described by use of these same modes. The experiment deployment modes that each cavity is capable of containing are presented in Table 3-1. Also shown in Table 3-1 are the possible directions in which an experimental payload can be ejected from the various cavities. These directions are measured from the particular cavity and are given in degrees up (Position III, + Z axis), down (I, -Z), left (IV, -Y), and right (II, + Y). All of the deployment modes except Modes 0 and 1 are designed for use with the vehicle configuration in which the Apollo payload has separated and the LEM Adapter Fairings are in the open position.

3.7 ENVIRONMENTS

3.7.1 Thermal

The thermal environment associated with each cavity is defined (Table 3-2) in terms of the maximum allowable rate of heat dissipation, the maximum total short-period heat dissipation, and the time-space averaged sink temperature. These parameters are dependent on the mission phase, and a separate specification is required for each phase - prelaunch, launch, and orbit. The temperature ranges shown in Table 3-2 represent variations of average values that are anticipated in the cavities.

TABLE 3-1
CAVITY POSSIBLE DEPLOYMENT MODES AND DIRECTIONS

• DIRECTION MEASURED FROM VEHICLE • FOR MODE 4 DEPLOYMENT ONLY
$+\theta$ UP, - DOWN $+\phi$ RIGHT, - LEFT

Cavity No.	Possible Deployment Modes	Direction (Degrees)	
		θ	ϕ
1-1	5, 4, 3, 2, 1, 0	+72 1/2 TO -72 1/2	+81 TO -35
1-2	4, 3, 2, 1, 0	+35 TO -81	+72 1/2 TO -72 1/2
1-3	5, 4, 3, 2, 1, 0	+72 1/2 TO -72 1/2	+35 TO -81
2-1	4, 3, 2, 1, 0	+64 TO -35	+56 TO -49 1/2
2-2	5, 4, 3, 2, 1, 0	+58 TO -58	+70 1/2 TO -35
2-3	4, 3, 2, 1, 0	+35 TO -64	+56 TO -49 1/2
2-4	4, 3, 2, 1, 0	+35 TO -64	+49 1/2 TO -56
2-5	5, 4, 3, 2, 1, 0	+58 TO -62	+35 TO -70 1/2
2-6	5, 4, 3, 2, 1, 0	+62 TO -58	+35 TO -70 1/2
2-7	4, 3, 2, 1, 0	+64 TO -35	+49 1/2 TO -56
3-1	4, 3, 2, 1, 0	+53 TO +9	+41 1/2 TO -41 1/2
3-2	4, 3, 2, 1, 0	+47 TO +9	+47 TO +9
3-3	4, 3, 2, 1, 0	+41 1/2 TO -41 1/2	+53 TO +9
3-4	4, 3, 2, 1, 0	-9 TO -47	+47 TO +9
3-5	4, 3, 2, 1, 0	-9 TO -54	+41 1/2 TO -41 1/2
3-6	4, 3, 2, 1, 0	-9 TO -47	-9 TO -47
3-7	4, 3, 2, 1, 0	+41 1/2 TO -41 1/2	-9 TO -53
3-8	4, 3, 2, 1, 0	+47 TO +9	-9 TO -47
4-1	4, 3, 2, 1, 0	+41 TO 0	+41 TO 0
4-2	4, 3, 2, 1, 0	0 TO -41	+41 TO 0
4-3	4, 3, 2, 1, 0	0 TO -41	0 TO -41
4-4	4, 3, 2, 1, 0	+41 TO 0	0 TO -41
Zones 5, 6 & 7	1, 0	0	0

TABLE 3-2
CAVITY THERMAL ENVIRONMENT

Parameter		Mission Phase		
		Prelaunch	Launch	Orbit
• MAX ALLOWABLE RATE OF HEAT DISSIPATION - ALL CAVITIES (BTU/HR)		200	100	300
• MAX TOTAL SHORT PERIOD HEAT DISSIPATION - ALL CAVITIES (BTU)		17	17	17
• TIME - SPACE AVERAGED TEMP (°F)				
CAVITIES	1-1 THRU 1-3	35 - 75	170 - 230	25 - 65
	2-1 THRU 2-7	35 - 75	200 - 240	-45 - 30
	3-1, 3-3, 3-5, & 3-7	35 - 75	180 - 220	35 - 65
	3-2, 3-4, 3-6, & 3-8	35 - 75	190 - 230	30 - 65
	4-1 THRU 4-4	35 - 75	25 - 65	25 - 65
	5-1 THRU 5-8	35 - 75	15 - 55	-105 - 55
	6-1 THRU 6-5	35 - 75	15 - 55	-105 - 55
	7-1 THRU 7-18	35 - 75	140 - 180	100 - 140

The maximum allowable rate of heat dissipation for all cavities varies from a maximum of 300 Btu per hour while in orbit to a minimum of 100 Btu per hour during launch. The maximum total short period heat dissipation for all cavities and all mission phases is 17 Btu. The temperatures vary from -105°F in Zone 5 and 6 cavities during orbit to +240°F in Zone 2 cavities during launch. These data are based on a ground rule that precludes the use of cooling air and equipment cold plates for control of heat dissipation from the experiment packages. In this analysis, then, it was assumed that the cavities contain no cold plate cooling or cooling air capacity.

3.7.2 Vibration and Acoustics

The vibration and acoustic environments associated with each cavity were defined in terms of sinusoidal vibration levels and a maximum overall sound pressure level as shown in Figure 3-7. The actual values used in this definition were extracted from design specifications contained in the Saturn IB Payload Planner's Guide (Reference 3-2) and represent the maximum environments to which components contained in these cavities would be subjected.

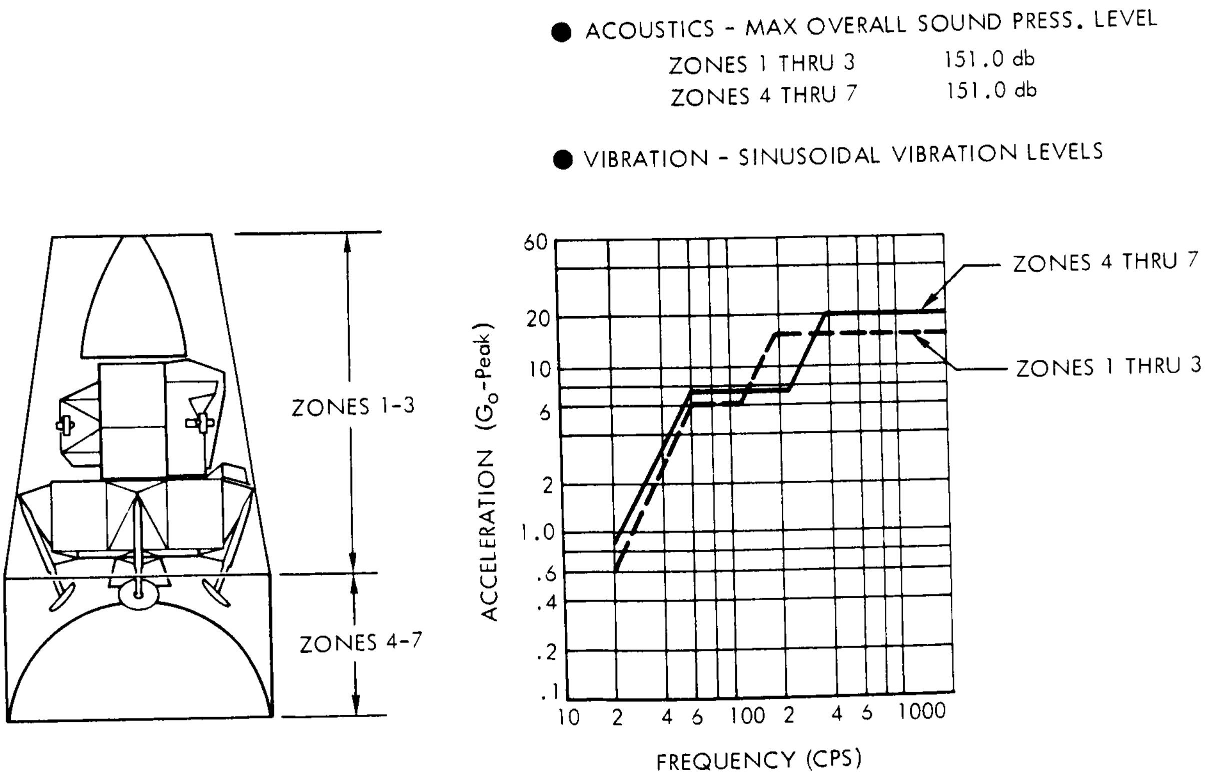


Figure 3-7 CAVITY VIBRATION AND ACOUSTICS ENVIRONMENT

As shown in Figure 3-7, the vibration level assigned to cavities in Zones 1 through 3 varies from 0.6-g at 20 cycles per second to 15-g at 200 cycles per second. The level for cavities in Zones 4 through 7 varies from 0.9-g at 20 cycles per second to 20-g at 400 cycles per second. The maximum overall sound pressure level of all cavities is 151.0 decibels.

3.7.3 Electromagnetic

Electromagnetic compatibility can be defined as the ability of each component in an integrated system to perform its design function without interfering with the performance of the design function of any other component in the system. The basic parameters which determine if one component will interfere with the function of another are

1. Level and bandwidth of signal a component is capable of emitting (transmitter signal)

2. Level and bandwidth of signal to which a component is capable of responding (receiver sensitivity)

3. Amount of isolation between components.

In order to limit the scope of this analysis it was necessary to assume that no isolation exists between the various experiments and that all cavities can be described by a vehicle electromagnetic environment. This environment is defined by a narrowband transmitter signal, a broadband transmitter signal, a narrowband receiver sensitivity, and a broadband receiver sensitivity as shown in Figure 3-8. The narrowband signal level and narrowband receiver sensitivity were obtained from various reports and specifications on the Saturn IB/V Instrument Unit. The broadband signal was based on emission limits outlined in the electromagnetic interference control specification MIL-I-6181D, and the broadband sensitivity was based on equipment functional test specifications. These values were obtained by a preliminary analysis and should be considered only as approximations made for the purpose of the computer program development and checkout.

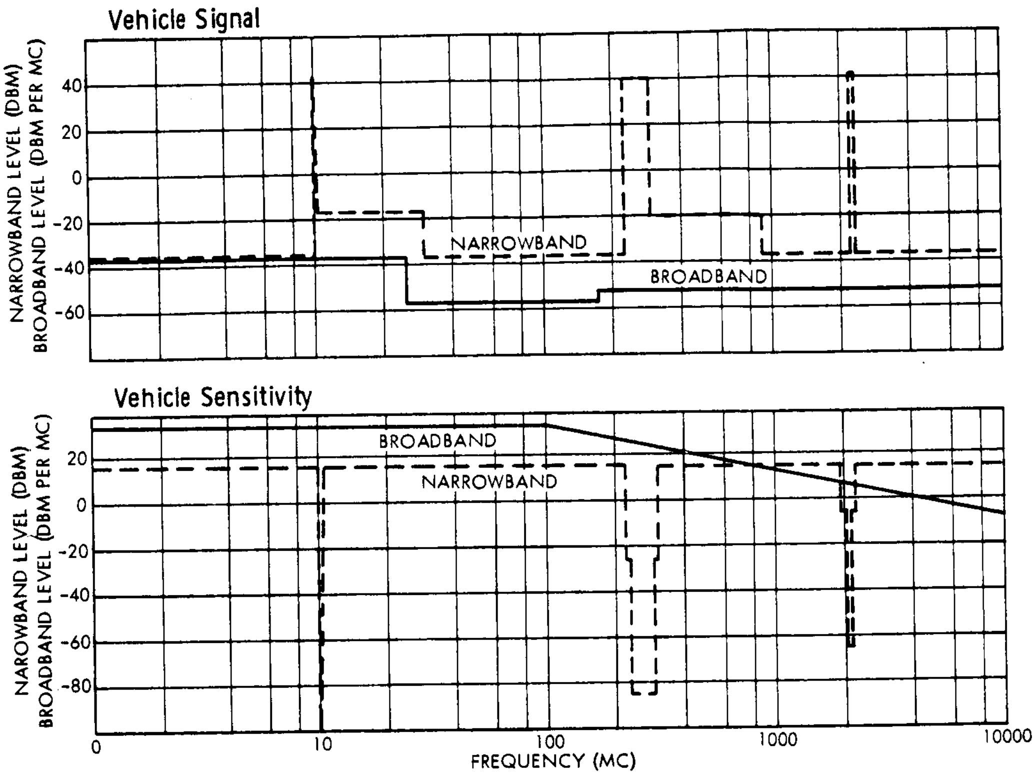


Figure 3-8 VEHICLE ELECTROMAGNETIC ENVIRONMENT

3.8 REFERENCES

3-1 Preliminary Definition of Saturn Instrument Unit and S-IVB Support Capabilities for Extended Apollo Earth-Orbit Experiments, National Aeronautics and Space Administration, George C. Marshall Space Flight Center, Huntsville, Alabama, 15 April 1965. (U)

3-2 Saturn IB Payload Planner's Guide, Douglas Report SM-47010, Missile and Space Systems Division, Douglas Aircraft Company, Inc., Santa Monica, California, March 1965. (U)

3-3 Apollo Spacecraft Structural Analysis of the SLA (Spacecraft LEM Adapter), North American Aviation, Inc., Space and Information Systems Division, 27 August 1965.(U)

SECTION 4

EXPERIMENT DESCRIPTION

METHODOLOGY

4.1 GENERAL

The experiment characteristics that are required for the payload characteristics library were developed in the experiment design effort which is presented in Volume II. The essential experiment characteristics that were obtained from this effort are summarized in this section. Those characteristics relating to experiment mass, deployment requirement, thermal environment, and electromagnetic environment can be used directly as inputs to the library. However, it was necessary to develop a specialized methodology for representing experiment volume/geometry and for providing vibration and acoustic mass penalties.

In the experiment design effort primary emphasis was placed on self-contained packages; that is, no consideration was given to the support capability of on-board equipment or to the sharing of subsystems with other experiments. However, to obtain a broader spectrum of data for use in the computer program checkout, the pertinent characteristics of certain vehicle-dependent experiments were also formulated. A vehicle-dependent experiment is defined as a self-contained experiment exclusive of power and communications subsystems and is indicated by an "A" after the basic experiment number. The experiments that were considered on both a self-contained and a vehicle-dependent basis are those 10 experiments which remain aboard the launch vehicle and are not ejected as separate satellites.

4.2 MASS

A summary of the masses of the self-contained and vehicle-dependent experiments is presented in Table 4-1. For those experiments which are ejected from the launch vehicle as separate satellites, two sets of data are given. The total installed mass is the total experiment mass installed on the launch vehicle. The total mass of separate satellite is the total mass of the satellite after separation from the launch vehicle.

4.3 VOLUME/GEOMETRY

To develop a method for representing experiment volume and geometry it was necessary to define two classes of experiments, fixed geometry and amorphous geometry, as shown in Figure 4-1. The fixed

TABLE 4-1
EXPERIMENT MASS SUMMARY

EXPERIMENT	TOTAL INSTALLED MASS (LBS)	TOTAL MASS OF SEPARATE SATELLITE (LBS)	EXPERIMENT	TOTAL INSTALLED MASS (LBS)	TOTAL MASS OF SEPARATE SATELLITE (LBS)
Self-Contained Experiments					
SDT-1	1031	937	SLG-1	191	--
SDT-2	58	53	SLG-2	195	--
SDT-3	682	636	SLG-3	713	648
SDT-4	487	443	SLG-4	201	--
SDT-5	440	400	SLG-5	153	--
MS-1	795	723	M-1	233	213
MS-2	103	--	M-2	165	--
MS-3	154	--	M-3	205	186
MS-4	173	--	M-4	135	123
MS-5	135	--	M-5	230	209
MI-1	1082	984	OEA-1	308	--
MI-2	896	815	OEA-2	312	284
MI-3	1434	1310	OEA-3	378	344
MI-4	798	730	OEA-4	401	365
MI-5	2812	2562	OEA-5	691	632
Vehicle-Dependent Experiments					
MS-2A	50	--	SLG-2A	86	--
MS-3A	60	--	SLG-4A	167	--
MS-4A	102	--	SLG-5A	69	--
MS-5A	74	--	M-2A	119	--
SLG-1A	160	--	OEA-1A	121	--

Fixed Geometry Experiments

Finalized Designs

DEFINED BY:

- STANDARD SHAPE ENVELOPE WHICH WILL CONTAIN THE ENTIRE EXPERIMENT

Amorphous Geometry Experiments

Amendable To Numerous Design Concepts

DEFINED BY:

- MINIMUM PRACTICAL INSTALLATION VOLUME &
- STANDARD SHAPE ENVELOPE WHICH WILL CONTAIN THE LARGEST UNDISTORTABLE COMPONENT

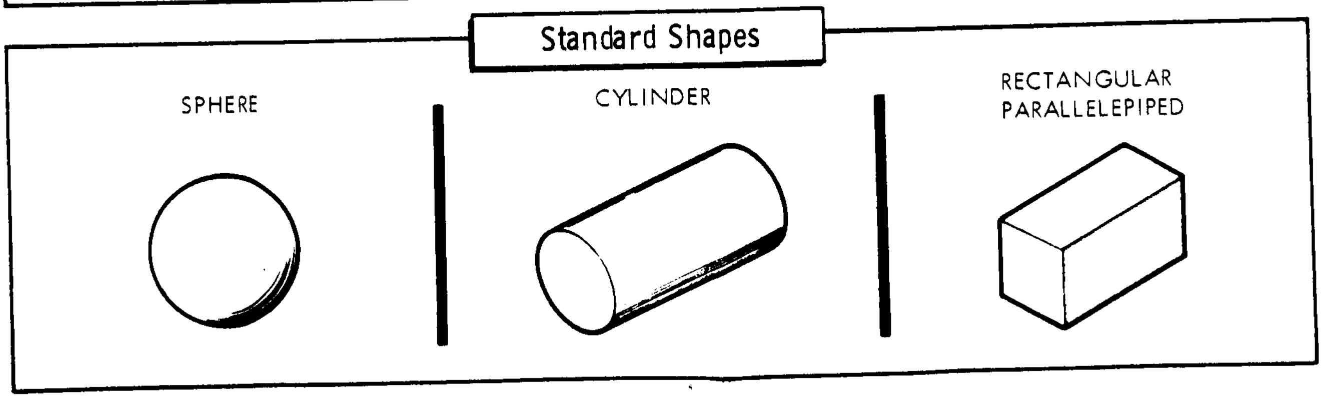


Figure 4-1 EXPERIMENT SHAPE AND VOLUME REPRESENTATION

geometry experiments are defined in terms of finalized designs whose geometry cannot be modified. The amorphous-geometry experiments are those in which the configuration is not fixed and which are amenable to numerous design concepts.

4.3.1 Fixed Geometry

The shape and volume of the fixed-geometry experiments is represented by a standard shape envelope which will most efficiently contain the entire experiment. The standard shapes selected for this representation are the sphere, the cylinder, and the rectangular parallelepiped. The energetic particles explorer satellite, experiment SDT-5, is shown in both the installed and the deployed configuration in Figure 4-2.

Fixed Experiments

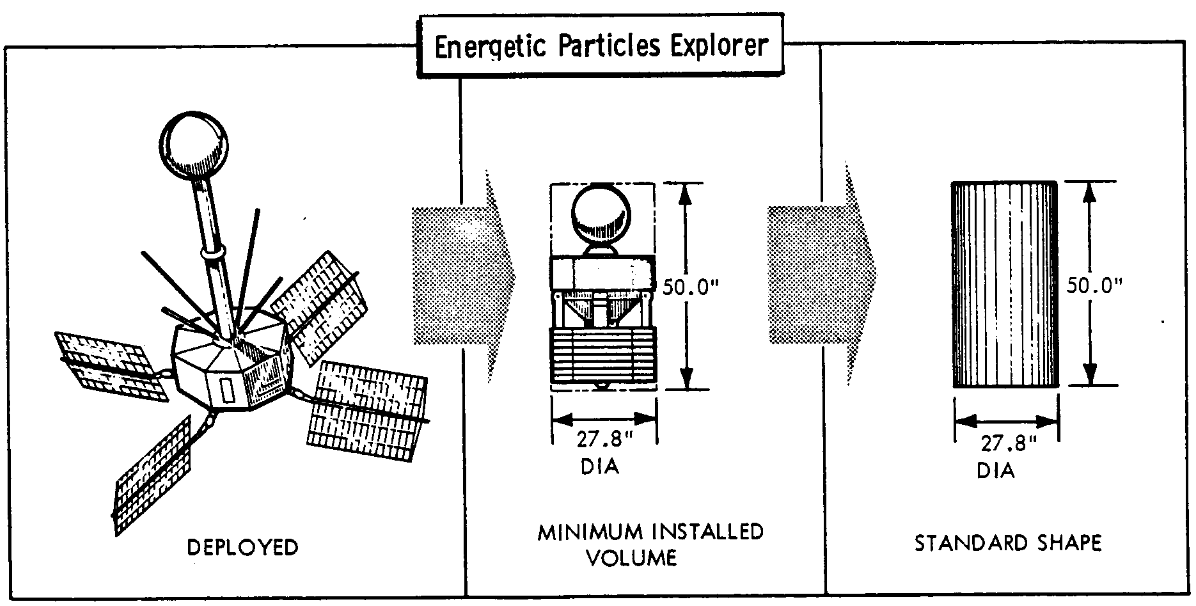


Figure 4-2 SHAPE AND VOLUME REQUIREMENTS - FIXED EXPERIMENTS

This is a designed hardware item and, therefore, represents fixed geometry. The shape and volume required for the installed configuration can most efficiently be represented by a standard shape cylinder which is 27.8 inches in diameter and 50.0 inches in length.

4.3.2 Amorphous Geometry

The shape of the amorphous-geometry experiments is represented by the standard shape envelope which will contain the experiment critical component. The critical component is an envelope of such size and shape that it will contain, in turn, each of the undistortable components in the experiment package. The critical component, then, can be either the largest undistortable component in the experiment or a composite of several undistortable components. The standard shapes used for this representation are the same as those used in the fixed experiments - sphere, cylinder, and rectangular parallelepiped. The volume of the amorphous geometry experiments is represented by a minimum volume for practical installation. For the self-contained experiments, this installation volume was obtained by multiplying the basic component volume by the applicable packaging factor. The packaging factor is the ratio of the installed volume to the basic component volue. A summary of the volumes, critical component sizes, and packaging factors for the representative experiments is presented in Table 4-2.

An example of the shape and volume representation of an amorphous geometry experiment is shown in Figure 4-3. The experiment, SLG-4,

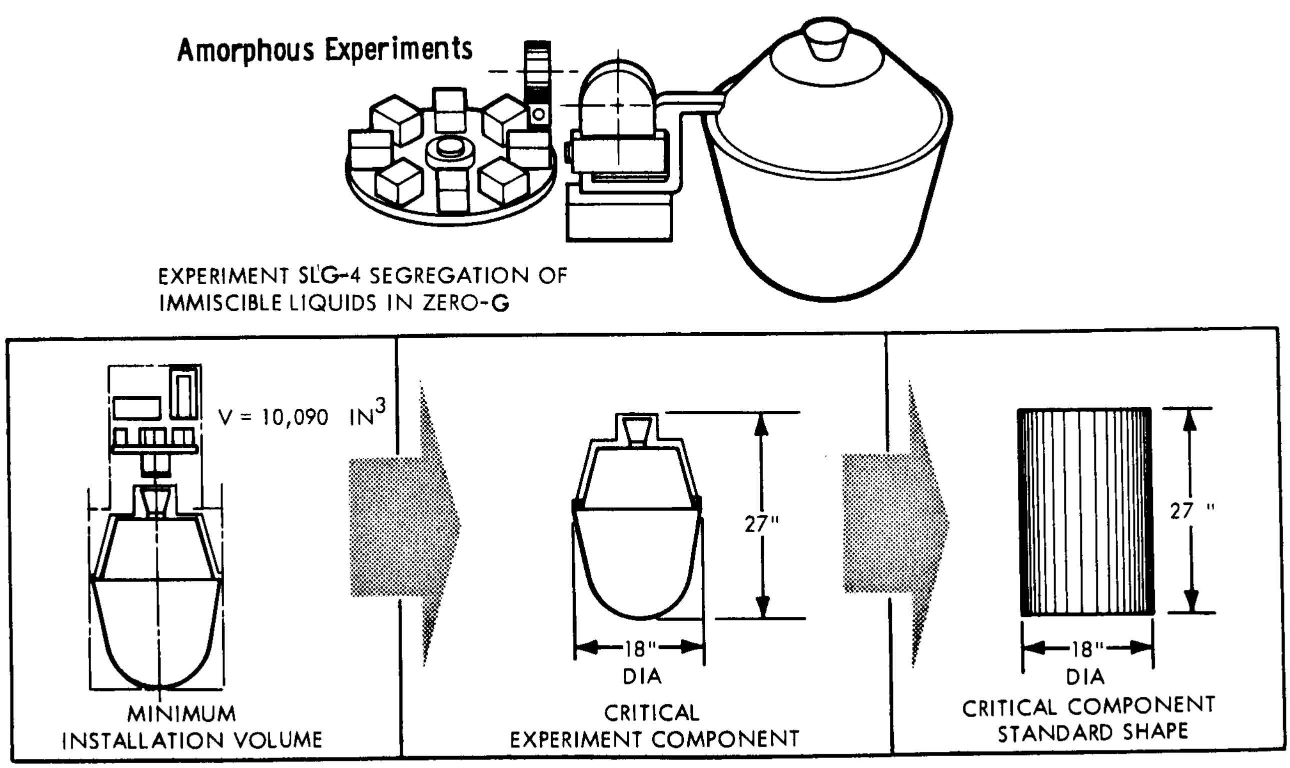


Figure 4-3 SHAPE AND VOLUME REQUIREMENTS - AMORPHOUS EXPERIMENTS

is composed of a group of components which can be arranged in a number of ways and still provide for a functional experiment. One possible arrangement is shown with the equipment mounted to the side of the

TABLE 4-2
EXPERIMENT VOLUME SUMMARY

EXPERIMENT	BASIC COMPONENT VOLUME (IN^3)	INSTALLED VOLUME (IN^3)	PACKAGING FACTOR	CRITICAL COMPONENT DIMENSIONS (IN)
Self-Contained Experiments				
SDT-1	50,900	122,000	2.40	30.0 x 30.0 x 24.0
SDT-2	2,504	3,689	1.47	6.0 x 6.0 x 36.0
SDT-3	25,803	72,900	2.83	21.0 DIA x 42.0
SDT-4	22,465	47,595	2.12	20.0 DIA x 48.0
SDT-5		30,349		27.8 DIA x 50.0
MS-1	46,200	73,500	1.59	36.0 DIA x 48.0
MS-2	1,827	2,988	1.64	10.0 x 12.0 x 12.0
MS-3	2,809	3,750	1.33	3.0 x 10.0 x 14.0
MS-4	4,344	5,500	1.27	4.0 x 8.5 x 14.0
MS-5	3,350	4,650	1.39	8.3 x 17.0 x 10.0
MI-1	31,670	81,200	2.56	44.0 x 22.0 x 31.0
MI-2	21,000	39,468	1.88	30.0 x 15.0 x 20.0
MI-3	98,000	139,000	1.42	12.0 x 60.0 x 80.0
MI-4	27,110	79,083	2.92	15.0 DIA x 51.0
MI-5	200,903	469,800	2.34	44.0 x 60.0 x 132.0
SLG-1	7,338	13,500	1.84	18.0 DIA x 25.0
SLG-2	3,450	7,862	2.28	17.0 x 11.0 x 6.0
SLG-3	43,950	102,993	2.34	33.5 DIA x 42.0
SLG-4	4,720	10,090	2.14	18.0 DIA x 25.0
SLG-5	5,130	7,100	1.38	10.0 x 10.0 x 13.0
M-1	7,427	17,400	2.34	14.0 DIA x 18.5
M-2	3,218	6,916	2.15	16.0 x 8.0 x 8.0
M-3	3,500	8,700	2.49	7.0 x 7.0 x 16.0
M-4	4,117	13,696	3.33	20.0 x 20.0 x 12.0
M-5	4,299	8,757	2.04	8.0 x 11.0 x 16.0
OEA-1	7,333	11,600	1.58	5.0 x 11.0 x 13.0
OEA-2	7,700	19,600	2.55	13.5 x 15.2 x 21.0
OEA-3	6,838	17,512	2.56	5.0 x 11.0 x 14.0
OEA-4	8,818	22,140	2.51	18.0 DIA x 25.0
OEA-5	12,456	31,870	2.56	15.0 DIA x 20.0
Vehicle-Dependent Experiments				
MS-2A	972	1,594	1.64	10.0 x 12.0 x 12.0
MS-3A	1,229	1,635	1.33	30.0 x 10.0 x 14.0
MS-4A	2,998	3,807	1.27	4.0 x 8.5 x 14.0
MS-5A	2,004	2,786	1.39	8.3 x 17.0 x 10.0
SLG-1A	6,914	12,722	1.84	18.0 DIA x 25.0
SLG-2A	1,620	3,694	2.28	17.0 x 11.0 x 6.0
SLG-4A	4,220	9,031	2.14	18.0 DIA x 25.0
SLG-5A	3,760	5,226	1.38	10.0 x 10.0 x 13.0
M-2A	2,488	5,349	2.15	16.0 x 8.0 x 8.0
OEA-1A	4,294	6,785	1.58	5.0 x 11.0 x 13.0

data recovery capsule. The minimum-installation-volume arrangement is shown with the equipment mounted above the data recovery capsule. Even though the experiment is considered as an amorphous configuration, there are some critical limiting dimensions to which the experiment can be conformed. In this experiment, the critical component is the data recovery capsule which is 18.0 inches in diameter and 27.0 inches in length. The standard shape which will most efficiently represent this critical component is a cylinder of the same dimensions.

4.4 DEPLOYMENT REQUIREMENT

During a mission, certain experiment requirements must be met in order to ensure the success of any experiment. Of particular interest in this study are those requirements that are contingent on proper installation of the experiment relative to the launch vehicle. These experiment requirements include exposure to vacuum, extension of an experiment component from the launch vehicle, and separation of a data recovery capsule. In order to fully describe these requirements for each experiment, six deployment modes, Mode 0 through Mode 5, were defined as shown in Figure 4-4. The modes were devised so that

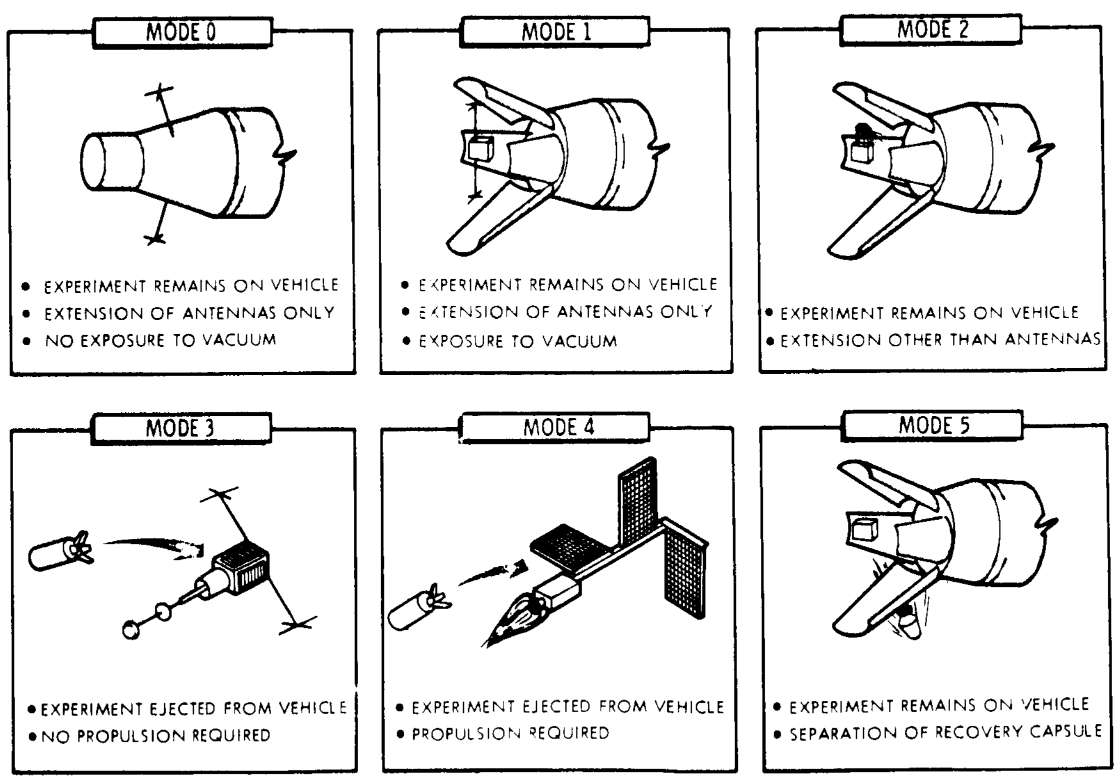


Figure 4-4 DEPLOYMENT MODE DEFINITION

it is necessary to use only one deployment mode to describe each experiment. A list of the thirty experiments and their required deployment modes is presented in Table 4-3.

TABLE 4-3
EXPERIMENT DEPLOYMENT MODES

Experiment	Mode	Experiment	Mode
SDT-1	3	SLG-1 & -1A	5
SDT-2	4	SLG-2 & -2A	0
SDT-3	3	SLG-3	3
SDT-4	3	SLG-4 & -4A	5
SDT-5	4	SLG-5 & -5A	0
MS-1	3	M-1	3
MS-2 & -2A	1	M-2 & -2A	0
MS-3 & -3A	1	M-3	3
MS-4 & -4A	1	M-4	3
MS-5 & -5A	2	M-5	3
MI-1	3	OEA-1 & -1A	2
MI-2	3	OEA-2	3
MI-3	3	OEA-3	4
MI-4	3	OEA-4	3
MI-5	3	OEA-5	3

4.5 ENVIRONMENTAL REQUIREMENTS

4.5.1 Thermal

The thermal environment for each of the experiments was defined by three parameters:

1. Maximum and minimum allowable time-space averaged temperature

2. Heat dissipation rate

3. Total short period heat dissipation.

As shown in Table 4-4, these parameters were determined for each of three mission phases: launch, prelaunch, and orbit. Since those experiments that are ejected from the spacecraft must be compatible with an orbital operational environment not associated with the spacecraft, no thermal compatibility checks will be made in the orbit mission phase for ejected experiments. The time-space averaged temperatures are the maximum and minimum temperatures to which the experiment components can be subjected without causing malfunctions.

TABLE 4-4
EXPERIMENT THERMAL ENVIRONMENT

Self-Contrained Experiments

EXPERIMENT	Prelaunch				Launch				Orbit			
	T_EMAX (°F)	T_EMIN (°F)	$\dot{Q}_E$ (BTU/HR)	Q_E (BTU)	T_EMAX (°F)	T_EMIN (°F)	$\dot{Q}_E$ (BTU/HR)	Q_E (BTU)	T_EMAX (°F)	T_EMIN (°F)	$\dot{Q}_E$ (BTU/HR)	Q_E (BTU)
SDT-1	100	0	0	0	250	0	0	0		EJECTED		
SDT-2	100	0	0	0	250	0	0	0		EJECTED		
SDT-3	80	35	0	0	250	35	0	0		EJECTED		
SDT-4	80	0	0	0	250	0	0	0		EJECTED		
SDT-5	100	0	0	0	250	0	0	0		EJECTED		
MS-1	75	35	0	0	240	35	0	0		EJECTED		
MS-2	75	14	27.3	N/A	260	0	27.3	N/A	65	-50	225	N/A
MS-3	75	14	27.3	N/A	240	0	27.3	N/A	65	-50	232	N/A
MS-4	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	191	N/A
MS-5	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	198	N/A
MI-1	80	30	0	0	300	30	0	0		EJECTED		
MI-2	100	20	0	0	400	20	0	0		EJECTED		
MI-3	100	0	0	0	400	0	0	0		EJECTED		
MI-4	100	0	0	0	400	0	0	0		EJECTED		
MI-5	90	10	0	0	400	10	0	0		EJECTED		
SLG-1	90	20	0	0	250	20	0	0	0	-50	1970	N/A
SLG-2	212	32	0	0	350	32	0	0	75	0	150	N/A
SLG-3	75	14	0	0	300	0	0	0		EJECTED		
SLG-4	75	35	0	0	250	35	0	0	65	20	392	N/A
SLG-5	75	14	0	0	250	0	0	0	75	-50	239	N/A
M-1	100	25	0	0	250	25	0	0		EJECTED		
M-2	80	0	17.1	N/A	250	0	17.1	N/A	80	0	17.1	N/A
M-3	80	0	20	N/A	250	0	20	N/A		EJECTED		
M-4	85	0	3.5	N/A	200	0	3.5	N/A		EJECTED		
M-5	90	0	3.5	N/A	250	0	3.5	N/A		EJECTED		
OEA-1	75	25	0	0	250	25	0	0	60	-50	394	N/A
OEA-2	100	0	0	0	200	0	0	0		EJECTED		
OEA-3	80	35	0	0	250	35	0	0		EJECTED		
OEA-4	75	35	0	0	250	35	0	0		EJECTED		
OEA-5	100	0	0	0	250	0	0	0		EJECTED		
Vehicle-Dependent Experiments												
MS-2A	75	14	27.3	N/A	260	0	27.3	N/A	65	-50	202	N/A
MS-3A	75	14	27.3	N/A	240	0	27.3	N/A	65	-50	196	N/A
MS-4A	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	181	N/A
MS-5A	75	14	27.3	N/A	250	0	27.3	N/A	65	-50	132	N/A
SLG-1A	90	20	0	0	250	20	0	0	0	-50	1960	N/A
SLG-2A	212	32	0	0	350	32	0	0	75	0	112	N/A
SLG-4A	75	35	0	0	250	35	0	0	65	20	392	N/A
SLG-5A	75	14	0	0	250	0	0	0	75	-50	172	N/A
M-2A	80	0	17.1	N/A	250	0	17.1	N/A	80	0	14	N/A
OEA-1A	75	25	0	0	250	25	0	0	60	-50	362	N/A

T_E - TIME-SPACE AVERAGED SINK TEMPERATURE
$\dot{Q}_E$ - HEAT DISSIPATION RATE
Q_E - TOTAL SHORT-PERIOD HEAT DISSIPATION

4.5.2 Vibration

Because the experiment vibration tolerance is very difficult to determine by analysis, the only meaningful vibration tolerance levels are those levels to which the experiment components have been qualified by testing. Two types of specification can be used in describing the vibration tolerance: random and sinusoidal. Because many off-the-shelf components have not been qualified to the random vibration specification, only sinusoidal vibration levels were considered in the compatibility checks. For the purposes of the computer program, a vibration tolerance level which would apply to the majority of off-the-shelf components was assigned to all experiments. This maximum sinusoidal vibration level, shown in Figure 4-5, is per MIL-E-5272C, Procedure XII.

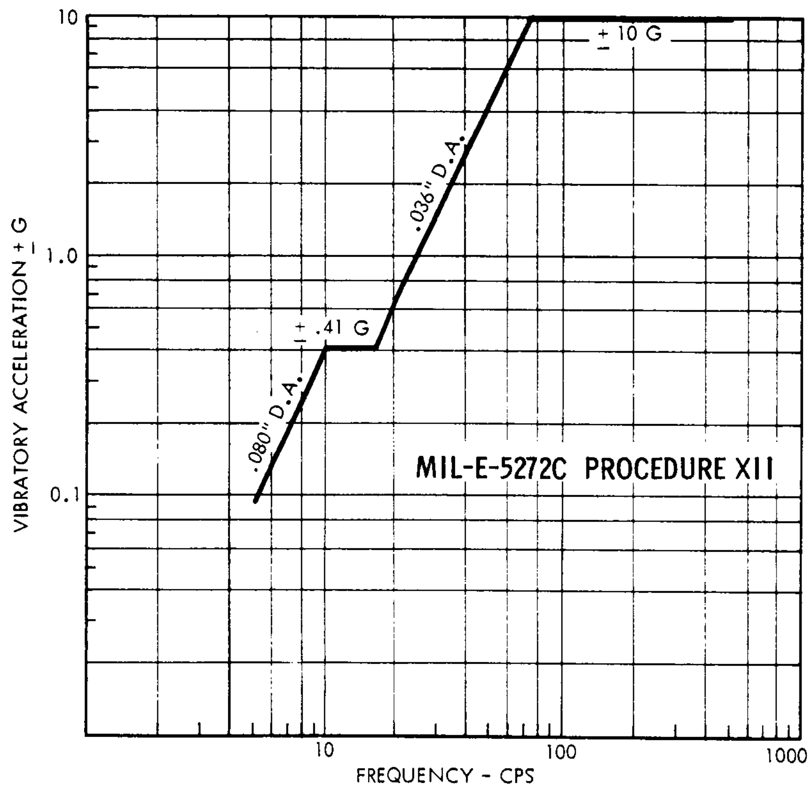


Figure 4-5 EXPERIMENT VIBRATION TOLERANCE

If in the compatibility check the experiment tolerance level were found to be below the environmental level this would result in a no-go evaluation. However, it is reasonable to assume that most, if not all, of the experiment components could be built to withstand the environmental vibration levels by the addition of mass to provide increased material gauges, isolation, stiffening, or damping. This mass penalty can then be assigned to experiments to increase their tolerance to a level that is compatible with the environment. Essentially, then, a mass penalty can be applied to change a NO-GO situation into a GO situation.

Several assumptions have been made in the determination of a technique to obtain this mass penalty. The launch environment is of the greatest significance since it is by far the most severe. It is assumed that the experiment components that are able to withstand the launch vibration will also be able to perform in the less severe environment of space. The vibration amplitude that a given material component can withstand is a measure of the component's strength which is, in turn, proportional to its dimensions or its mass. The amplitude of vibration tolerance is seen by the foregoing to be proportional to the mass of the affected parts. The experiment packages contain components of varying degrees of vibration failure susceptibility. Some components by nature are not susceptible to vibration, hence should not enter into the calculation of the mass penalty for vibration tolerance deficiency. The structure of the experiment package and mounting brackets is placed in this category by the assumption that the detail structural design is adequate for the expected environments.

The only available quantity which can be used as a measure of the varying susceptibility of the remaining components is the density (mass per unit volume) of the experiment package. Generally the relative density of a component is a measure of its vibration tolerance. The following equation, which was developed by a rough dimensional analysis, yields the mass penalty factor for an experiment:

$$\Delta W_V = \frac{g}{g_1} \times \frac{D_r}{D_e} \times W_V$$

where W_V is the mass penalty for vibration tolerance deficiency in pounds

g_1 is the vibratory acceleration amplitude of the assigned vibration tolerance

Δg is the difference between the vibration acceleration amplitude of the desired vibration tolerance and the assigned vibration tolerance

D_r is a reference density (lb/in.^{3})

D_e is the experiment density (lb/in.^{3})

W_V is the mass of the vibration susceptible components of the experiment package.

The penalty weight is sensitive to the choice of the reference density (D_r). Unfortunately, its choice must be arbitrary at this time. Reasoning that the percentage increase in weight required should not exceed half of the percentage increase in vibration tolerance required, a value of half the density of the least dense experiment of the current experiment list was chosen as the reference density. The weight penalty for vibration tolerance deficiency, then, ranges from a few pounds for the most dense experiment to a maximum of one-half the percentage increase in g-level times the weight of the sensitive components in the least dense experiment. Table 4-5 is a summary of the mass of the vibration and noise susceptible components of the reference experiments.

TABLE 4-5
MASS SUMMARY VIBRATION/NOISE SUSCEPTIBLE COMPONENTS

Self - Contained Experiments

Experiment	Mass (Lbs)	Experiment	Mass (Lbs)	Experiment	Mass (Lbs)
SDT-1	291	MI-1	130	M-1	82
SDT-2	25	MI-2	301	M-2	128
SDT-3	324	MI-3	607	M-3	72
SDT-4	80	MI-4	325	M-4	62
SDT-5	-	MI-5	465	M-5	65
MS-1	71	SLG-1	67	OEA-1	95
MS-2	68	SLG-2	68	OEA-2	156
MS-3	63	SLG-3	159	OEA-3	105
MS-4	91	SLG-4	71	OEA-4	73
MS-5	60	SLG-5	64	OEA-5	224
Vehicle - Dependent Experiments					
MS-2A	50	SLG-1A	160	SLG-5A	69
MS-3A	60	SLG-2A	86	M-2A	119
MS-4A	102	SLG-4A	167	OEA-1A	121
MS-5A	74				

4.5.3 Acoustics

The same difficulty is encountered in defining the acoustical noise tolerance of the experiments as was encountered in defining the vibration tolerance. The tolerance levels assigned to the experiments can only be as high as the levels to which the experiment components have been qualified by testing. Because off-the-shelf components were used, whenever possible, in the experiment definitions, a maximum noise tolerance of 150 db overall was assigned to all experiments. This value is per MIL Std 810 "Acoustical Test Method, Grade B."

If in the compatibility check, the experiment noise tolerance was less than the environmental level, it would be necessary to apply a mass penalty factor, as was done with vibration incompatibility, to change the NO-GO evaluation to a GO evaluation. The same general reasoning that was used for the relationship of vibration tolerance, the launch environment, and the experiment characteristics can be used for the noise problem. The environmental change factor, however, must be stated in terms of sound pressure levels rather than g levels. The susceptibility index or experiment mass factor and the weight of the sensitive components, Table 4-5, remain the same. The weight penalty estimate for noise tolerance deficiency can be written:

$$W_n = \frac{\Delta SPL}{SPL_1 - 140\ db} \times \frac{D_r}{D_e} \times W_n$$

Where:

ΔSPL is the difference between the desired noise tolerance and the assigned noise tolerance.

SPL_1 is the sound pressure level of the assigned noise tolerance

140 db is the threshold of acoustic noise damage

$\frac{D_r}{D_e}$ is unchanged from the vibration problem

W_n is the weight of the noise susceptible components.

The techniques that improve noise tolerance usually improve vibration tolerance and vice versa. However, since the range of frequencies associated with each are different, the mass used in solving either a vibration or an acoustic problem can not be considered as solving the other also.

4.5.4 Electromagnetic

Electromagnetic compatibility can be defined as the ability of each component in an integrated system to perform its design function without interfering with the performance of the design function of any other component in the system. The basic parameters which determine if one component will interfere with the function of another are eneumerated below:

1. Level and bandwidth of signal which a component is capable of emitting (transmitter signal).

2. Level and bandwidth of signal to which a component is capable of responding (receiver sensitivity)

3. "Coincident time interval" or the occurrence of simultaneous operation of components whose parameters, (1) and (2) above, overlap

4. Amount of isolation between components.

To provide for an electromagnetic compatibility check between the experiments and the launch vehicle, the emission spectrum and the receiver sensitivity were defined for the selected experiments. Because of the isolation provided by the distance between the ejected experiments and the launch vehicle, only the ten experiments that remain aboard the launch vehicle were analyzed for the electromagnetic compatibility parameters. As shown in Figure 4-6 and Table 4-6, each experiment is described by a narrowband transmitter signal, a broadband transmitter signal, a narrowband receiver sensitivity, and a broadband receiver sensitivity.

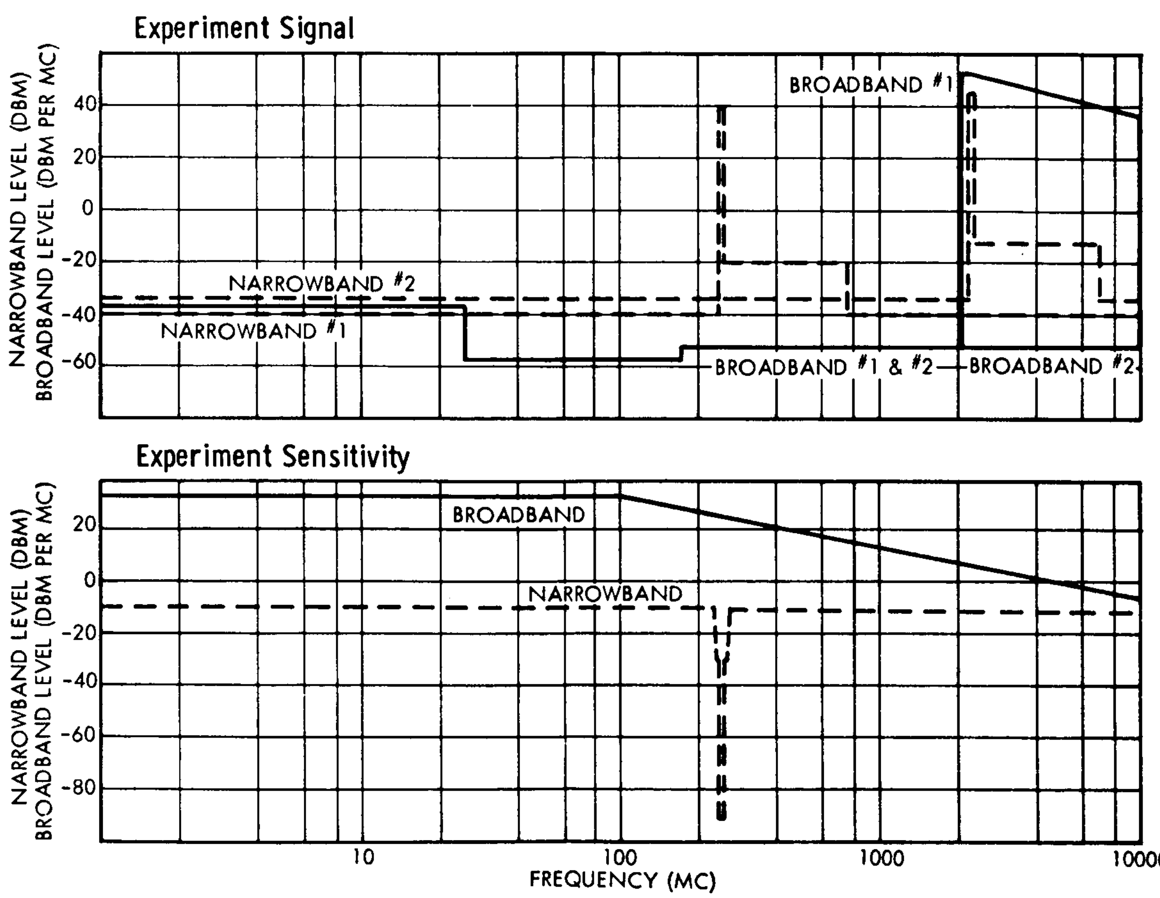


Figure 4-6 EXPERIMENT ELECTROMAGNETIC ENVIRONMENT

TABLE 4-6
EXPERIMENTAL ELECTROMAGNETIC ENVIRONMENT

Applicable Curves	Experiments									
	MS -2	MS -3	MS -4	MS -5	SLG -1	SLG -2	SLG -4	SLG -5	M -2	OEA -1
NARROWBAND SIGNAL #1	√	√	√	√	√	√		√		√
NARROWBAND SIGNAL #2									√	
BROADBAND SIGNAL #1	√	√	√	√		√		√	√	
BROADBAND SIGNAL #2					√		√			√
NARROWBAND SENSITIVITY	√	√	√	√	√	√		√	√	√
BROADBAND SENSITIVITY	√	√	√	√	√	√	√	√	√	√

SECTION 5

EXPERIMENT/MISSION EFFECTIVENESS ANALYSIS

5.1 EXTERNAL ANALYSIS APPROACH

Experiment effectiveness is defined as the percent of accomplishment of data acquisition objectives. For the type of experimental payloads considered in this study (i.e., specified components), effectiveness is primarily a function of the initial elements and other parameters of the deployed orbit (experiment/mission effectivensss). Other factors (e.g., payload location, angular rates, reliability, etc.) are recognized as potentially significant, but generally secondary, influences. The determination of experiment effectiveness as a function of the initial orbital elements and mission parameters is accomplished by the use of methods (including auxiliary computer procedures) which are not included in Program SEPTER.

5.1.1 Data Acquisition Objectives

Each experiment is analyzed in order to define the data acquisition objectives. Basic effectiveness definitions are formulated as functions of the parameters which affect experiment effectiveness. The basic effectiveness parameters are, in general, not restricted to orbital or mission parameters. Environmental parameters, for example, may directly affect the data acquisition objectives. In this case, experiment effectiveness must first be expressed in terms of these parameters which are in turn related to the initial orbital elements and/or mission parameters.

The accomplishment of the complete data acquisition objectives of a scientific program may require more than a single flight of the experiment. In this case, the maximum effectiveness attainable for one flight of the experiment will be less than 100 percent even if an optimum orbit is flown.

5.1.2 Trajectory/Mission Analysis

Experiment effectiveness is computed in Program SEPTER by multiplying effectiveness factors which are functions of one or more of the orbital elements and/or mission parameters listed in Table 5-1. Effectiveness factor functions are derived from the basic effectiveness definitions after trajectory/mission analyses have been performed to establish the effectiveness relationships. For some experiments,

TABLE 5-1
ORBITAL ELEMENTS AND MISSION PARAMETERS

VARIABLE IDENTIFICATION NUMBER	VARIABLE	UNITS
1	Semi-major Axis	km
2	Eccentricity	-
3	Inclination	deg
4	Argument of Perigee	deg
5	True Anomaly	deg
6	Time of Perigee Passage	sec
7	Perigee Latitude	deg
8	Period	sec
9	Apogee Altitude	km
10	Perigee Altitude	km
11	Apogee/Perigee Altitude Ratio	-
12	Longitude of Nodal Passage	deg
13	Time of Nodal Passage	sec
14	Inclination to Terminator	deg
15	Solar Declination	deg
16	Launch Month	-
17	Launch Year	-
18	Julian Date of Launch	-
19	Duration of Primary Mission	days
20	Launch Time	hrs

the experimenter and the mission analyst may have to perform a rather extensive analysis in order to arrive at meaningful effectiveness relationships.

The trajectory/mission data necessary to relate the experiment effectiveness to the initial orbital elements and/or mission parameters are generated by the use of auxiliary computer procedures. Efficient computation of atmospheric perturbations is essential for the analysis of orbital decay effects. Analytical approximation formulae such as those of King Hele (Reference 5-1) are adequate for predicting orbital lifetimes greater than a few days.

5.2 EXAMPLE EFFECTIVENESS ANALYSES

The data acquisition objectives, the flight regime, and the operational aspects of each of the 30 selected experiments were analyzed to identify the orbital elements and mission parameters which influence experiment effectiveness. As a result of this analysis, the 20 variables listed in Table 5-1 were selected for use in computing experiment effectiveness in Program SEPTER. The important flight regimes and operations characteristics of representative experiments (one from each of the six experiment categories) are summarized in Table 5-2. Experiment effectiveness of each experiment can be expressed as a function of the initial orbital elements and/or mission parameters shown in the last column.

Effectiveness analyses were conducted for 20 selected experiments for the purpose of developing an external analysis approach and illustrating representative effectiveness relationships. Basic effectiveness definitions and final effectiveness relationships for these experiments are contained in Appendix B.

The extent and complexity of the required analyses were found to vary considerably between the various candidate experiments. In order to demonstrate this variation and to illustrate the actual analyses which must be performed to obtain effectiveness data, the analyses for the six representative experiments listed in Table 5-2 are presented on the following pages.

The experiment design orbit is defined as the initial orbit which yields maximum effectiveness. In some cases, the experiment effectiveness of the design orbit is less than 100 percent. This implies that no single orbit can be used to attain all the data acquisition objectives of the scientific program. In other experiments, 100 percent effectiveness may be readily obtained in more than one orbit.

TABLE 5-2
FLIGHT REGIMES AND OPERATIONS CHARACTERISTICS OF REPRESENTATIVE EXPERIMENTS

IDENTIFICATION		TITLE	FLIGHT REGIME	OPERATIONS					ORBITAL ELEMENTS/MISSION PARAMETERS AFFECTING EFFECTIVENESS
CATEGORY	NO.			DURATION	DUTY CYCLE	ATTITUDE	DATA RATE	DEPLOYMENT MODE	
SDT	4	CRYOGENIC PROPELLANT STORAGE SYSTEM PERFORMANCE	NO RESTRICTIONS	4 DAYS	CONTINUOUS AFTER INJECTION	TOWARD SUN $\pm 3^{\circ}$	500-1000 BITS ORBIT	SEPARATION	• PERIGEE ALTITUDE • APOGEE ALTITUDE • INCLINATION TO TERMINATOR
MS	3	VAPORIZATION RATE OF MOLTEN METALS	• APPROXIMATEL CIRCULAR ORBIT • PERIGEE $>$ 148 KM	29 HRS	0.3 TO 4 HR RUNS	LOW TUMBLING RATE	26-HR TAPE	EXTENSION	• PERIGEE ALTITUDE • APOGEE ALTITUDE
MI	1	MULTI-SPECTRAL SURVEILLANCE OF EARTH	• ALTITUDE RANGE $<$370 KM • SUN ANGLE 30° ABOVE HOR • INCLINATION DEPENDS ON TARGET	14 DAYS	40 1 MIN RUNS	UP TO $\pm 30^{\circ}$ FROM VERTICAL $\pm 1.5^{\circ}$ POINTING ACCURACY	DATA EVERY 3 SECONDS	SEPARATION	• PERIGEE ALTITUDE • APOGEE ALTITUDE • INCLINATION • INCLINATION TO TERMINATOR • PERIGEE LATITUDE • SOLAR DECLINATION
SLG	2	NUCLEATE CONDENSATION IN ZERO-GRAVITY	• ACCELERATION $<$ .01 g's	24 HRS	ONE 24 HR CYCLE	NO CONTROL	30 MINUTES OF TAPE ORBIT	• EXTENSION • PROPULSIVE SEPARATION	• PERIGEE ALTITUDE • APOGEE ALTITUDE
M	5	PRODUCTION OF NUTRIENTS BY CERTAIN MICRO-ORGANISMS WHILE IN SPACE FLIGHT	NO RESTRICTIONS	15 DAYS	1 HR CYCLE, 15-20 MINUTES RUN	NO CONTROL	30-100 READINGS EVERY 2 DAYS	SEPARATION	• PERIGEE ALTITUDE • APOGEE ALTITUDE
OEA	2	STUDY OF MAGNETIC FIELD LINES	• ALTITUDE $<$ 370 KM • INCLINATION $\approx 90^{\circ}$	50 ORBITS	15 MIN RUN 50 RUNS	ALONG FIELD LINES $\pm 5^{\circ}$	4000 BITS RUN	SEPARATION	• PERIGEE ALTITUDE • APOGEE ALTITUDE • INCLINATION • SEMIMAJOR AXIS

Finally, the effectiveness data are prepared for inclusion in the effectiveness segment of the Experimental Payload Characteristics Library.

5.2.1 Category I Experiments - Systems Development and Testing

Category I experiments involve the development and testing of advanced subsystems, techniques, and processes for the support of future space operations. Orbital elements and mission parameters important to the successful accomplishment of the data acquisition objectives of each experiment of Category I are included in the following:

1. Perigee altitude

2. Apogee/perigee altitude ratio

3. Inclination to the equatorial plane

4. Inclination to the terminator plane.

Experiment SDT-2 is deployed from the vehicle with a propulsive ΔV (Deployment Mode 4); SDT-3 and SDT-4 are deployed without propulsive ΔV (Deployment Mode 3). Experiment SDT-5, a "fixed-design" experiment, is deployed without propulsive ΔV into a parking orbit prior to the initiation of a three-impulse transfer maneuver into a high altitude, elliptic orbit. This deployment requirement corresponds to a Mode 4 deployment (i.e., the orbit of the experiment is modified from the launch vehicle orbit by a propulsive ΔV). Since effectiveness is more conveniently defined in terms of the orbital elements of the parking orbit, a Deployment Mode 3 was assumed for effectiveness computation. Experiments SDT-1, SDT-3, and SDT-4 are attitude controlled.

Experiment SDT-4, "Cryogenic Propellant Storage System Performance" is a typical representative of the experiments in Category I. The objectives of experiment SDT-4 are to (1) evaluate the performance of certain thermal protection systems, (2) determine the degree of propellant stratification, and (3) evaluate the performance of an ullage orientation system for the reduction of propellant stratification.

The effectiveness of SDT-4 is dependent upon four parameters: (1) useful orbital lifetime, (2) mean drag acceleration of the initial orbit, (3) change in mean drag acceleration over the mission duration, and (4) initial inclination to the terminator. The first three parameters are determined by the atmospheric decay of orbit altitude and can be expressed in terms of the initial perigee and apogee altitude.

Thus, the experiment effectiveness can be defined in terms of the initial perigee altitude, the initial apogee/perigee altitude ratio, and the initial inclination of the orbit to the terminator plane.

Experiment effectiveness can be expressed as the product of the timing factor E_t, the initial mean acceleration factor E_a, the mean acceleration change factor $E_{\Delta a}$, and the inclination to terminator factor E_{F_2}. The basic effectiveness factor relationships (Fig. 5-1) were established by the experimenter after an analysis of the effects of mission duration and drag acceleration on experiment effectiveness.

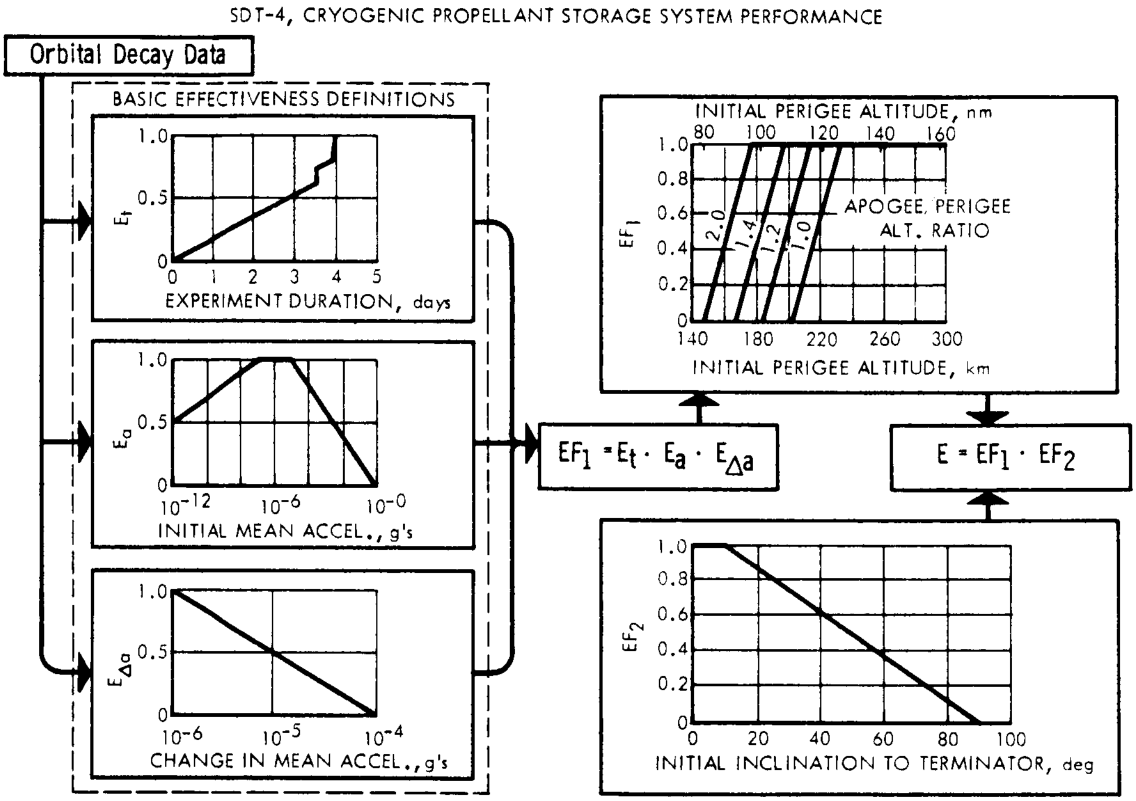


Figure 5-1 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS SDT-4

5.2.2 Category II Experiments - Materials and Structures

The objectives of the Category II experiments are the test and evaluation of materials and structures which are exposed to the space environment. Orbital elements and mission parameters important to the successful accomplishment of the data acquisition objectives of Category II experiments include the following:

1. Perigee altitude

2. Apogee/perigee altitude ratio

3. Inclination to terminator (MS-1 only).

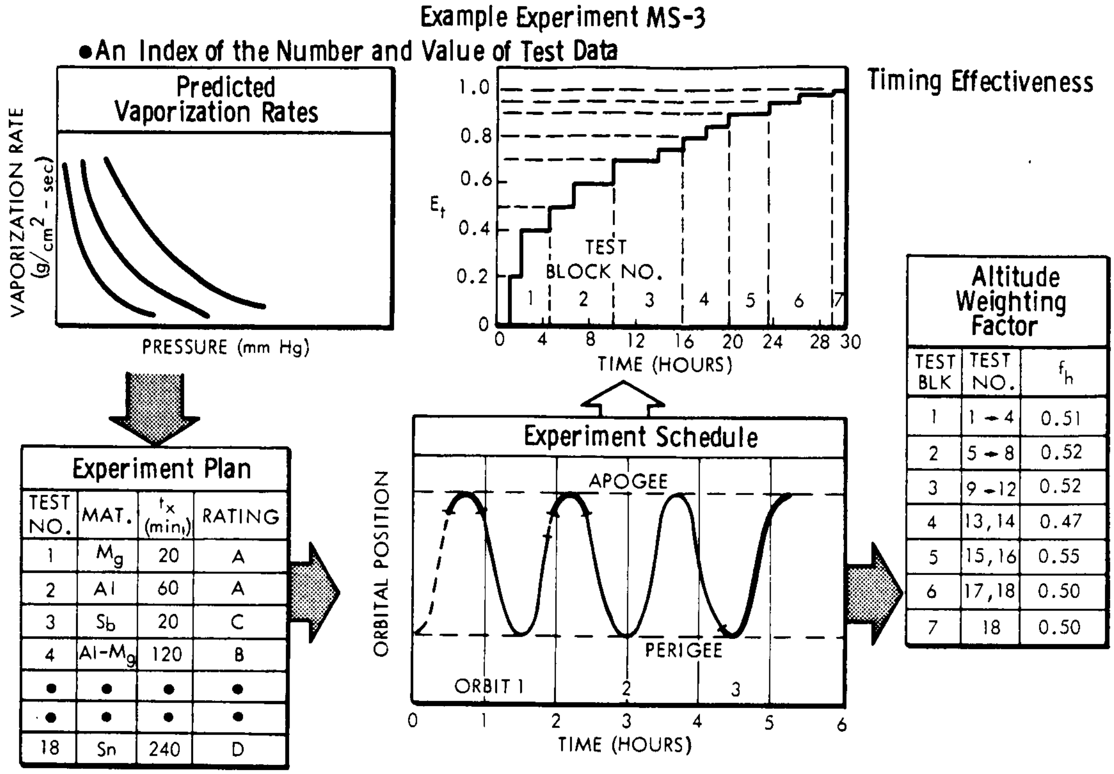
With the exception of MS-1, all experiments in this category are retained on the launch vehicle in a space environment (Deployment Mode 1 or 2). Experiment MS-1 is deployed from the vehicle without propulsive ΔV and is attitude controlled to maintain a solar orientation.

An illustration of one of the more complex analyses required to obtain effectiveness relationships is provided by an examination of Experiment MS-3, "Vaporization Rate of Molten Metals." The data acquisition objective set for this experiment is to determine the vaporization characteristics of various molten metals in near-Earth orbits. The data accumulated will be used to verify the predicted (calculated from theory) vaporization rates of metals in a very low atmospheric pressure environment. To obtain 100 percent effectiveness, the experimenter specified that the atmospheric pressure was not to exceed 10^{-7} millimeters of Hg for the duration of the experiment (29 hours). The basic effectiveness variables are, therefore, atmosphereic pressure and time.

To accomplish the data acquisition objective, the experimenter planned 18 vaporization tests, using nine selected materials. Each material was rated on the basis of the scientific and practical value of the data it could yield. In the experiment plan of Figure 5-2, ratings A, B, C, etc., designate decreasing yields. On the basis of this rating and the predicted vaporization rates, the individual experiments were then scheduled in such a way that the amount and value of test data would decrease with time. The experiment schedule is partially illustrated in Figure 5-2.

Utilizing the experiment schedule, the experimenter subjectively evaluated the data yield as a function of experiment duration to arrive at a "timing effectiveness" function $E_t(t)$ as shown in the upper right graph of Figure 5-2. The timing effectiveness factor E_t is an index to the amount and value of the test data accumulated at any time in the experiment. The experimenter also assigned altitude "weighting factors," f_h, to each of several blocks of tests, as shown in the listing at the extreme right in Figure 5-2. These factors were used for computing the "effective" altitude (as defined in Figure 5-3) of each block of tests. In the case of short tests, which would occur near apogee, the factor is nearly 1.0; consequently, the effective altitude is weighted toward the average apogee altitude.

Because of the time variance of altitude and, therefore, atmospheric pressure, the concept of effective altitude was introduced



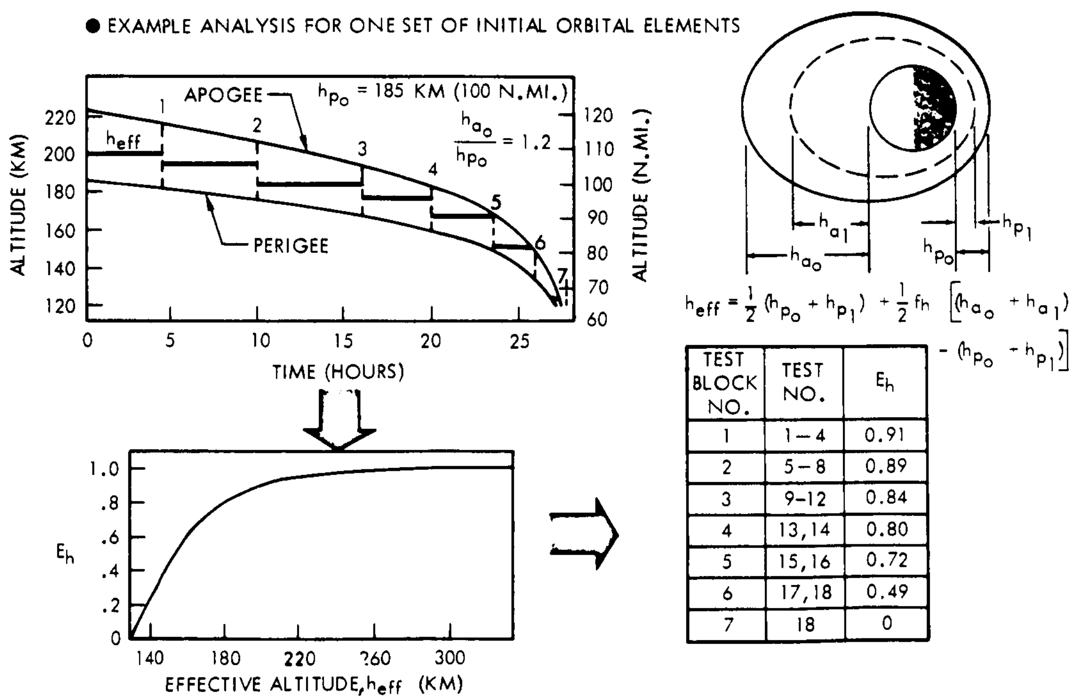
Experiment Plan

TEST NO.	MAT.	t_x (min.)	RATING
1	Mg	20	A
2	Al	60	A
3	Sb	20	C
4	Al-Mg	120	B
•	•	•	•
•	•	•	•
18	Sn	240	D

Altitude Weighting Factor

TEST BLK	TEST NO.	f_h
1	1 → 4	0.51
2	5 → 8	0.52
3	9 → 12	0.52
4	13, 14	0.47
5	15, 16	0.55
6	17, 18	0.50
7	18	0.50

Figure 5-2 DETERMINATION OF TIMING EFFECTIVENESS FACTOR EXAMPLE EXPERIMENT MS-3



$$h_{eff} = \frac{1}{2}(h_{p_o} + h_{p_1}) + \frac{1}{2} f_h \left[(h_{a_o} + h_{a_1}) - (h_{p_o} - h_{p_1})\right]$$

TEST BLOCK NO.	TEST NO.	E_h
1	1—4	0.91
2	5—8	0.89
3	9—12	0.84
4	13, 14	0.80
5	15, 16	0.72
6	17, 18	0.49
7	18	0

Figure 5-3 DETERMINATION OF ALTITUDE EFFECTIVENESS FACTOR EXAMPLE EXPERIMENT MS-3

to simplify the experiment effectiveness analysis. For a given test, the effective altitude was determined by adding the weighted average of the difference between the perigee and apogee altitudes to the average perigee altitude (see equation in Figure 5-3). The weighting factor f_h in the equation was derived subjectively by the experimenter. In the example experiment, significant decay of perigee and apogee altitude occurred during the experiment. Therefore, the tests were divided into blocks, and the effective altitude was determined for each block. After the effective altitude was determined, the altitude effectiveness factor E_h was defined subjectively as a function of the calculated effective altitude. This relationship is given in the lower left graph in Figure 5-3.

A sample analysis of an orbit with an initial perigee altitude of 185 kilometers (100 nautical miles) and an initial apogee/perigee altitude ratio of 1.2 is shown in Figure 5-3. The experiment was divided into seven test blocks, and the effective altitude was computed for each block. From the altitude effectiveness curve, E_h was determined and tabulated for each test block. From the tabulated data in Figure 5-3, it can be seen that the altitude effectiveness decreases rapidly after test 16 and becomes zero in the last test. The procedure illustrated in Figure 5-3 was repeated for a matrix of initial perigee altitudes and apogee/perigee altitude ratios to complete this portion of the analysis.

The final step in this example analysis was to compute the experiment effectiveness from the timing and altitude effectiveness factors, E_t and E_h. This computation was done by multiplying the sum of the product $(\Delta E_t \cdot E_h)$ by an eccentricity factor f_e. The quantity ΔE_t is the change in E_t over the duration of the test block, and E_h is the altitude effectiveness of the test block. The eccentricity factors were used to adjust the effectiveness relationship for a slight degradation of the data caused by the altitude variation that results from orbital eccentricity.

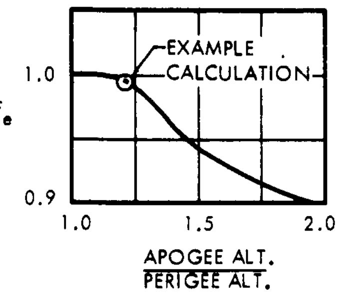
Experiment effectiveness is shown in Figure 5-4 as a function of the initial orbital elements (i.e., perigee altitude, and apogee/perigee altitude ratio). These are the data that are required for the effectiveness segment of the Experimental Payload Characteristics Library.

The design orbit for the example experiment is an initially circular orbit at an altitude of 333 kilometers (180 nautical miles). In this case, the maximum effectiveness for the design orbit is 100 percent. Because of the eccentricity factor, 100 percent effectiveness can be achieved only in circular orbits.

Example Experiment MS-3

$$E = \left(\sum_{1}^{n} \Delta E_t \times E_h\right) f_e = 0.84 \text{ For Example Calculation}$$

$$\Delta E_t = E_{t_n} - E_{t_{n-1}}$$



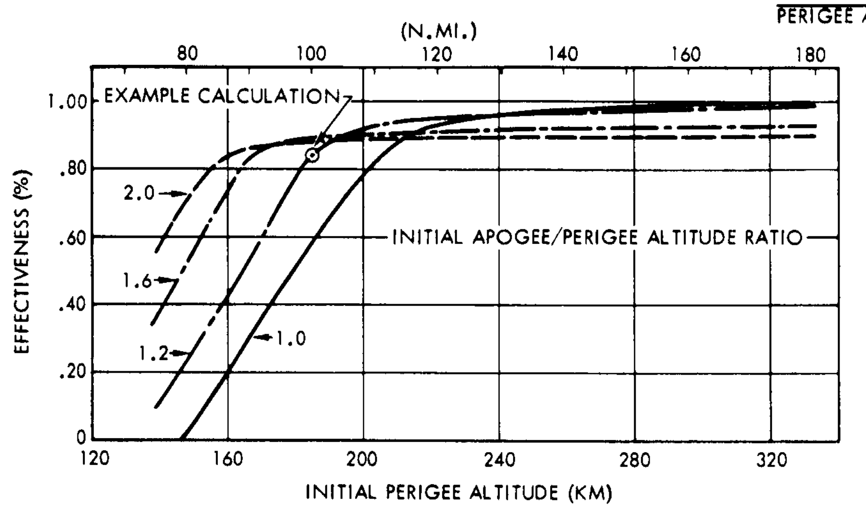


Figure 5-4 EFFECTIVENESS AS A FUNCTION OF INITIAL ORBITAL ELEMENTS

5.2.3 Category III Experiments - Multispectral Imagery of the Earth and Orbiting Objects

Category III experiments involve the collection of multispectral data of selected areas of the Earth's surface to permit the analysis of geographic and geologic features, agricultural and economic resources, and meteorological and oceanographic conditions. Orbital elements and mission parameters important to the successfull accomplishment of the data acquisition objectives of Category III experiments include the following:

1. Perigee altitude

2. Apogee/perigee altitude ratio

3. Inclination to equatorial plane

4. Inclination to terminator

5. Perigee latitude

6. Declination of sun

7. Julian date (MI-5 only)

All selected experiments in this category are deployed from the launch vehicle without propulsive ΔV (Deployment Mode 3) and are attitude controlled to provide Earth orientation.

Experiment MI-1, "Multispectral Surveillance of Earth," is typical of the experiments in Category III. The data acquisition objective of MI-1 is to obtain simultaneously a set of aerial photographs and spectral radiometric data (in various bands of the visible and infrared spectrum) of selected areas of the Earth's surface. The data will be returned to Earth via data recovery capsules.

Prior to flight a number of target areas which are accessible to ground inspection will be selected. Since areas in the United States will be of particular interest because of their accessibility, most of the data runs will be performed over the United States.

Experiment effectiveness of MI-1 is primarily a function of seven parameters: (1) experiment duration, (2) perigee altitude, (3) perigee latitude, (4) apogee/perigee altitude ratio, (5) inclination to equatorial plane, (6) solar declination, and (7) inclination to the terminator.

The "timing effectiveness" function $E_t(t)$ is shown in Figure 5-5. The timing effectiveness factor E_t is an index to the amount and value of test data accumulated at any time in the duration of the experiment. Since experiment duration can be expressed as a function of the initial values of perigee altitude and apogee/perigee altitude ratio, the timing E_t can likewise be expressed in terms of these orbital elements.

The experimenter specified that over land areas data runs should be performed at low altitude (167 km for 100% effectiveness) and that altitudes above 370 km would not be acceptable. An altitude effectiveness factor E_h was defined to evaluate the degradation of the data due to excessive altitude. The factor E_h is shown in Figure 5-5 as a function of perigee altitude and apogee/perigee altitude ratio.

The importance of the inclination of the orbit to the equatorial plane will depend to a large extent on the areas selected for observation. Inclinations greater than 48 degrees (or less than 132 degrees) afford complete coverage of the United States and are defined as having an inclination factor E_i equal to 1.0. Inclinations outside this region have lower values of E_i as shown in Figure 5-5.

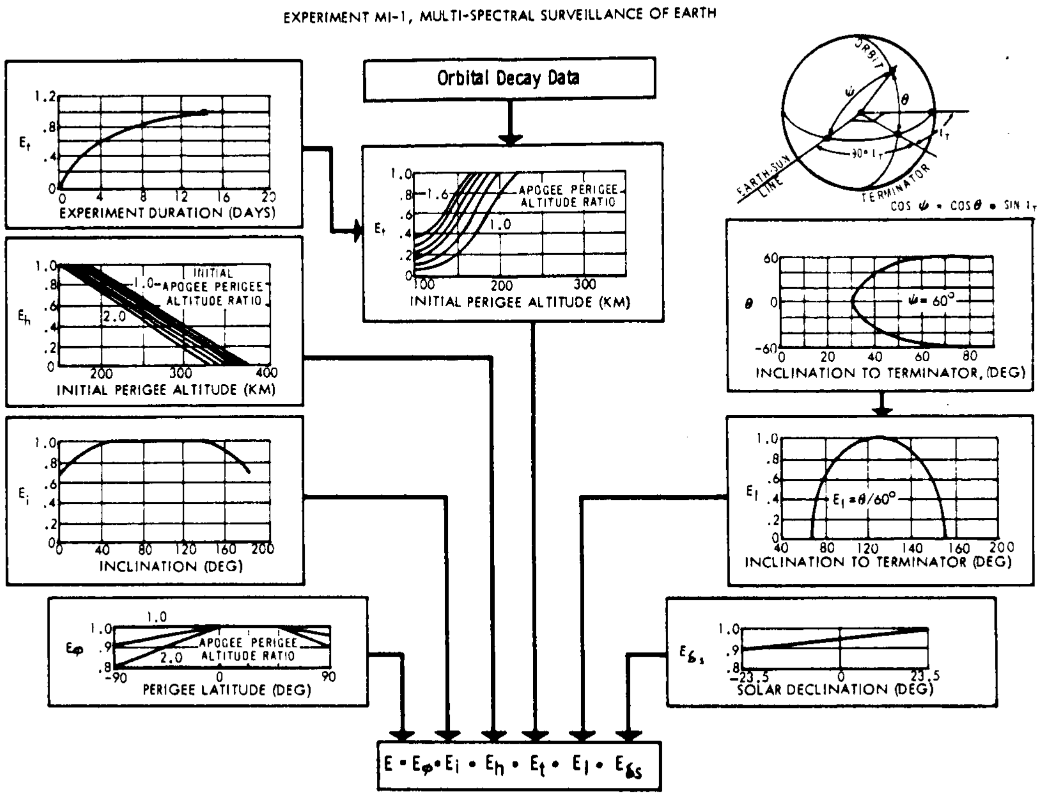


Figure 5-5 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS MI-1

The primary sensors (photographic cameras) require daylight illumination. The illumination angle at any point on the ground track can be determined from the inclination to the terminator by the relationship

$$\cos \psi = \cos \theta \sin I_T$$

where ψ is the zenith angle of the sun at the point, I_T is the inclination to the terminator, and θ is the central angle defined in Figure 5-5. Since the experimenter specified that zenith angles greater than 60 degrees were unacceptable, the fraction of the ground track which is properly illuminated is given by

$$F = \frac{2\theta}{2\pi} = \frac{1}{\pi} \cos^{-1} \left[\frac{\cos \pi/3}{\sin I_T} \right] \qquad \frac{2\pi}{3} \geqslant I_T \geqslant \frac{\pi}{6}$$

The illumination factor E_I, defined as F/F_{max}, is shown in Figure 5-5 as a function of inclination to the terminator, I_T. Because of the importance of observations over the United States, a slight degradation of effectiveness occurs for eccentric orbits when the latitude of perigee is less than 23 degrees or greater than 48 degrees.

This effect becomes more pronounced with eccentricity (i.e., apogee/perigee ratio) and is accounted for by subjectively defining a perigee latitude factor $E\phi$ shown in Figure 5-5. Similarly, the declination of the sun will have some influence on the illumination angle for targets located in the United States and is accounted for by a solar declination factor given by

$$E_{\delta_S} = 0.95 + 0.00213\ \delta_S.$$

The experiment effectiveness data are loaded into the effectiveness array of the Experimental Payload Characteristics Library in six tables. Effectiveness is then computed in SEPTER by multiplication of the six factors:

$$E = E_t \cdot E_h \cdot E_i \cdot E_\phi \cdot E_I \cdot E_{\delta S}.$$

The design orbit for Experiment MI-1 is a circular orbit with an initial inclination to the equator greater than 48 degrees and an initial inclination to the terminator of 90 degrees.

5.2.4 Category IV Experiments - Solid/Liquid/Gas Behavior

Category IV experiments are designed to provide data on the behavior and characteristics of liquids, solids and gases in zero-g environment. With the exception of experiment SLG-3 the experiments in this category require only a low acceleration environment for the duration of the experiment to accomplish all data acquisition objectives. Experiment SLG-3 requires, in addition, continuous exposure to direct sunlight (twilight orbit) to achieve 100 percent effectiveness. The orbital elements and mission parameters which influence the effectiveness of experiments in Category IV include the following:

1. Perigee altitude

2. Apogee/perigee altitude ratio

3. Inclination to the terminator (SLG-3 only).

All experiments in this category are retained on the launch vehicle except Experiment SLG-3 which is deployed without propulsive ΔV (Deployment Mode 3). In Experiments SLG-1 and SLG-4, a data capsule is ejected after completion of the experiment (Deployment Mode 5). Experiment SLG-3 is attitude-controlled to maintain a solar orientation during the experiment and to orient the experiment vehicle for data capsule ejection at the completion of the experiment.

Experiment SLG-2, "Nucleate Condensation in Zero Gravity," is typical of the Category IV experiments. The objective of Experiment SLG-2 is to observe nucleate condensation in a zero-gravity environment. Data are recorded on magnetic tape and relayed back at 4-hour intervals. In order to achieve 100 percent effectiveness, the drag acceleration must be less than 0.01 g for the duration of the experiment (24 hours). Should the experiment be terminated during one of the 4-hour test intervals, the data recorded in that interval are considered unavailable. Therefore, the effectiveness variation with experiment duration is a series of step functions as shown in the upper left graph in Figure 5-6.

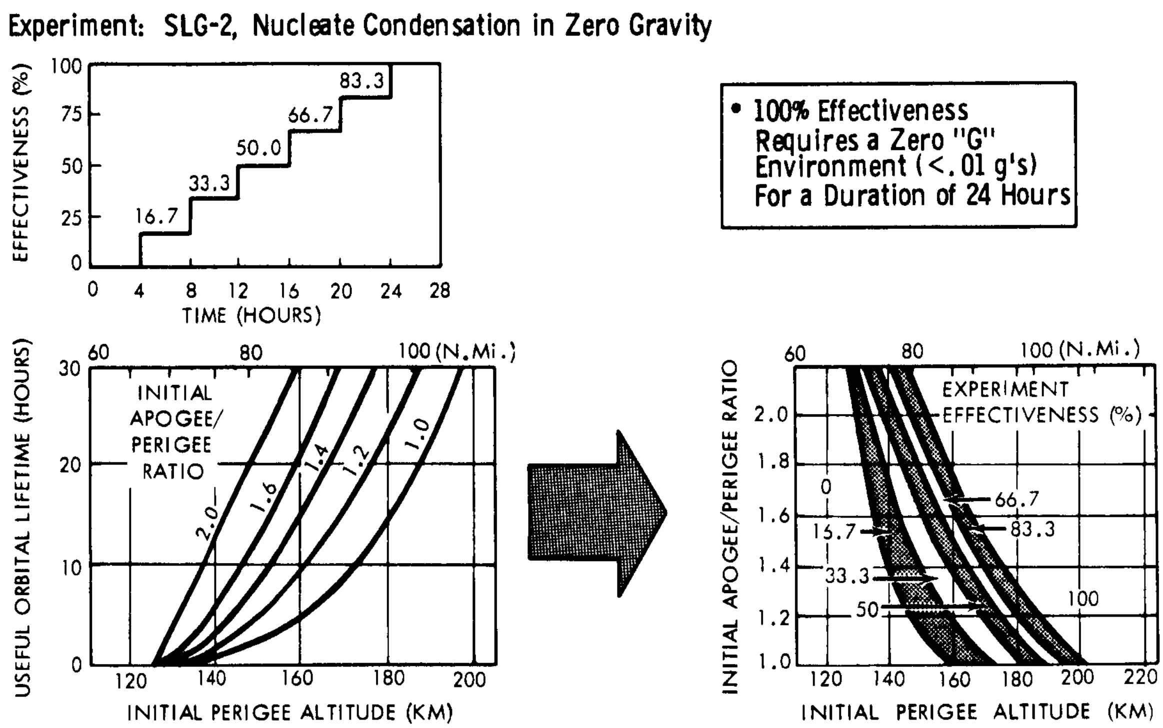


Figure 5-6 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS SLG-2

To relate experiment effectiveness to the initial perigee and apogee altitudes, the useful orbital lifetime was determined as a function of these elements (lower left graph). The useful orbital lifetime for this experiment is defined as the period of time when the drag acceleration is less than 0.01 g (i.e., perigee is less than about 104 kilometers). From the lifetime data, curves of perigee altitude versus apogee/perigee ratio were generated for 4-, 8-, 12-, 16-, 20-, and 24-hour orbital lifetimes corresponding to 16.7-, 33.3-, 50-, 66.7-, 83.3-, and 100 percent effectiveness values, respectively.

Any initial set of perigee and apogee altitudes which provide an orbital lifetime of 24 hours will have an effectiveness of 100 percent as shown in the lower right graph in Figure 5-6.

5.2.5 Category V Experiments - Microorganisms

Category V experiments were designed to provide data on the effects of space flight on various microorganisms. Additionally, experiment M-1 involves the soft capture and enumeration of spaceborne microorganisms. To accomplish all data acquisition objectives, the experiments in this category must remain at orbital altitudes for the duration of the experiment. Experiment effectiveness is, therefore, a function of perigee altitude and apogee/perigee attitude ratio (i.e., orbital lifetime). All experiments in this category are deployed from the launch vehicle without propulsive ΔV (Deployment Mode 3) with the exception of Experiment M-2 which is retained aboard the launch vehicle (Deployment Mode 0). Experiment M-1 is attitude-controlled to maintain an orientation along the velocity vector; Experiments M-3, M-4, and M-5 are not attitude controlled.

Experiment M-5, "Production of Nutrients by Certain Microorganisms While in Spaceflight," is typical of the experiments in Category V. The objective of Experiment M-5 is to demonstrate the effects of extended space flight on the production of nutrients by microorganisms. Turbidometric measurements of the growth of nutrient precursor-dependent mutants of selected bacteria and mutants requiring the nutrient will be transmitted to earth hourly from the deployed experiment. To achieve all data acquisition objectives, the measurements must be completed for a time period of 15 days. The variation of experiment effectiveness with experiment duration is shown in Figure 5-7. After the first two hours, when no data are collected, the effectiveness factor increases with time and becomes one after a period of 15 days. The effectiveness is expressed as a function of initial perigee and apogee/perigee altitude ratio (i.e., orbital lifetime) as shown in Figure 5-7 for inclusion in the effectiveness array of the Experimental Payload Characteristics Library.

5.2.6 Category VI Experiments - Observation of the Earth's Atmosphere, the Space Environment and Astronomical Phenomena

Category VI experiments involve observation of the Earth's atmosphere and magnetic field geometry, measurements of the space environment, and astronomical observations. Orbital elements and mission parameters important to the accomplishment of the data acquisition objectives of Category VI experiments include the following:

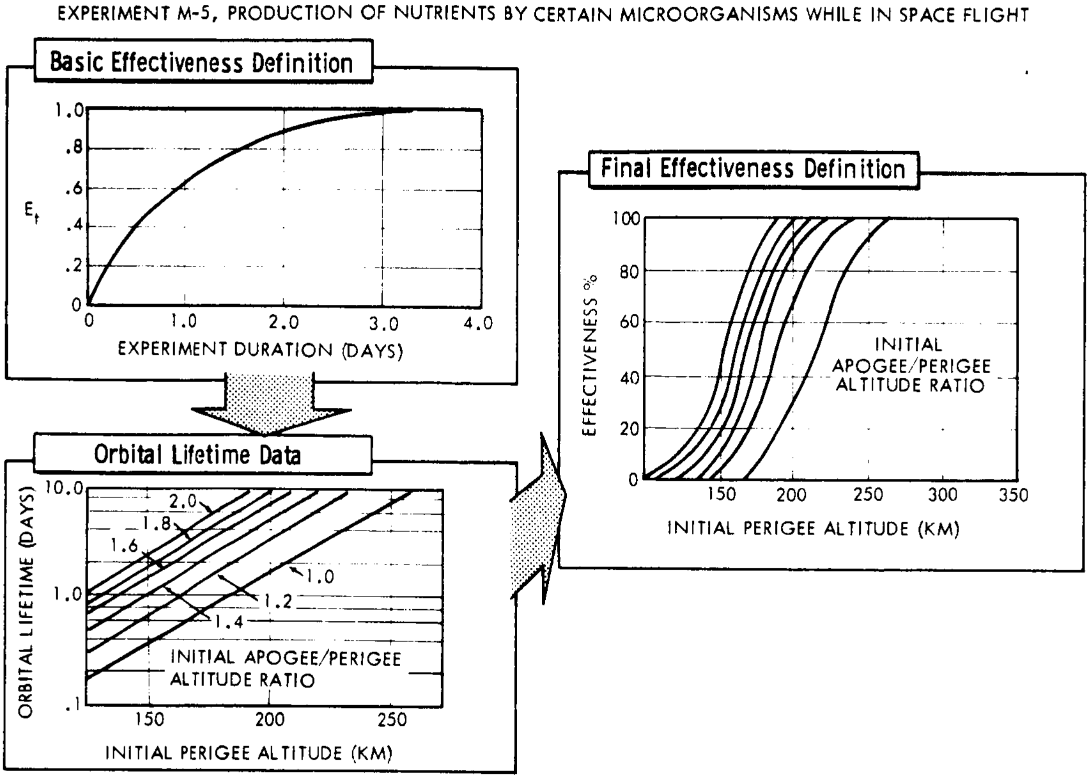


Figure 5-7 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS M-5

1. Perigee altitude
2. Apogee/perigee altitude ratio
3. Apogee altitude
4. Inclination to the equatorial plane.

With the exception of Experiments OEA-1 and OEA-4, the experiments in this category are deployed from the launch vehicle without propulsive ΔV (Deployment Mode 3). Experiment OEA-4 is deployed with propulsive ΔV (Deployment Mode 4). Experiment OEA-1 is unique in that the experiment is divided into two components: (1) radiation monitoring equipment to record the external radiation environment and (2) radiation monitoring equipment to record the radiation received by the crew. The external radiation monitor is retained with the launch vehicle while the internal radiation monitor is located inside the crew compartment. A Deployment Mode 3 is specified for this experiment.

With the exception of Experiment OEA-1, the experiments in this category are attitude controlled.

Experiment OEA-2, "Study of Magnetic Field Lines," is typical of the experiments in Category VI. The objective of this experiment is to observe and study the geometry of the lines of force of the terrestrial magnetic field by measuring the paths followed by electrons artificially injected along the field lines. When these electrons contact the atmosphere, an auroral spot is produced which can be tracked visibly by motion picture photography or radar from stations other than the spacecraft. To achieve all data acquisition objectives the deployed orbit must cross all field lines, avoid interference from the Earth's natural radiation belts, and have a useful lifetime of 50 orbits. Optimal altitudes for electron ejection were specified by the experimenter to be 370.4 kilometers (200 nautical miles). Effectiveness is therefore a function of experiment duration, apogee and perigee altitudes, and orbital inclination.

The timing effectiveness function $E_t(N)$ shown in Figure 5-8 was defined subjectively by the experimenter. The factor E_t was then

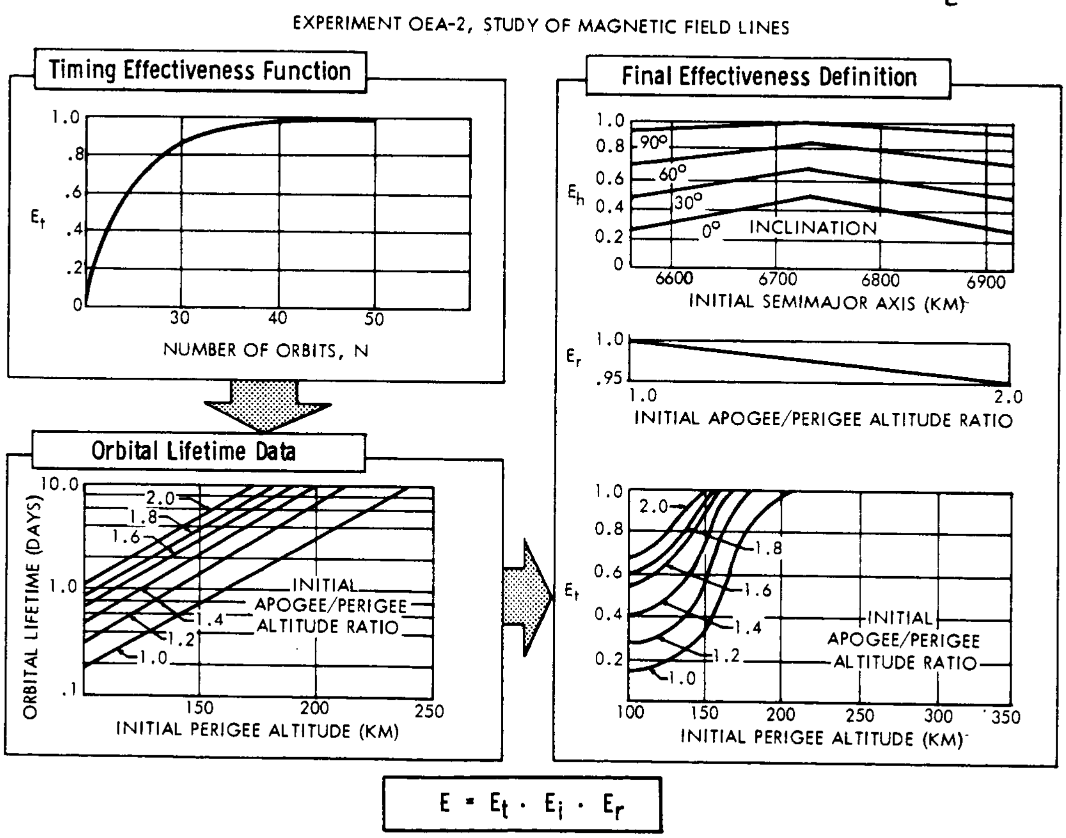


Figure 5-8 EXAMPLE EXPERIMENT EFFECTIVENESS ANALYSIS OEA-2

related to the initial perigee altitude and apogee/perigee altitude ratio (i.e., orbital lifetime) by assuming a nominal orbital period of 1.5 hours. This relationship is shown in Figure 5-8. An altitude/inclination function $E_h(a,i)$ for circular orbits was also defined subjectively by the experimenter as shown in Figure 5-8 as a function of inclination and semi-major axis. This factor degrades experiment effectiveness for off-optimum altitude and/or inclination conditions. For elliptic orbits the factor E_h was expressed as a function of the "average" altitude (average of apogee and perigee) and a third effectiveness function $E_R(R)$ was defined to account for a reduction in experiment effectiveness resulting from altitude variations (i.e., eccentricity). The altitude ratio factor is given by the relationship

$$E_R = 1.05 - 0.05\ R$$

where R is the initial apogee/perigee altitude ratio.

The data shown in Figure 5-8 were loaded into the effectiveness array of the Experimental Payload Characteristics Library. Experiment effectiveness is then computed as the product of the three effectiveness factors:

$$E = E_t \cdot E_h \cdot E_R.$$

The design orbit is a 370.4 kilometer (200 nautical mile) circular orbit with an inclination of 90 degrees.

5.3 REFERENCES

5.1 King-Hele, Desmond, Theory of Satellite Orbits in an Atmosphere, Butterworths, 1964. (U)

part II PROGRAM SEPTER

SECTION 6

PROGRAM SEPTER PHILOSOPHY AND LOGIC

6.1 GENERAL CONSTRUCTION AND OPERATION

The overall construction and flow of calculations in Program SEPTER are shown schematically in Figure 6-1. Included are the types and forms of data inputs, the major areas of analyses (designated as subroutines), and the types and forms of output data for each mode of operation.

SEPTER

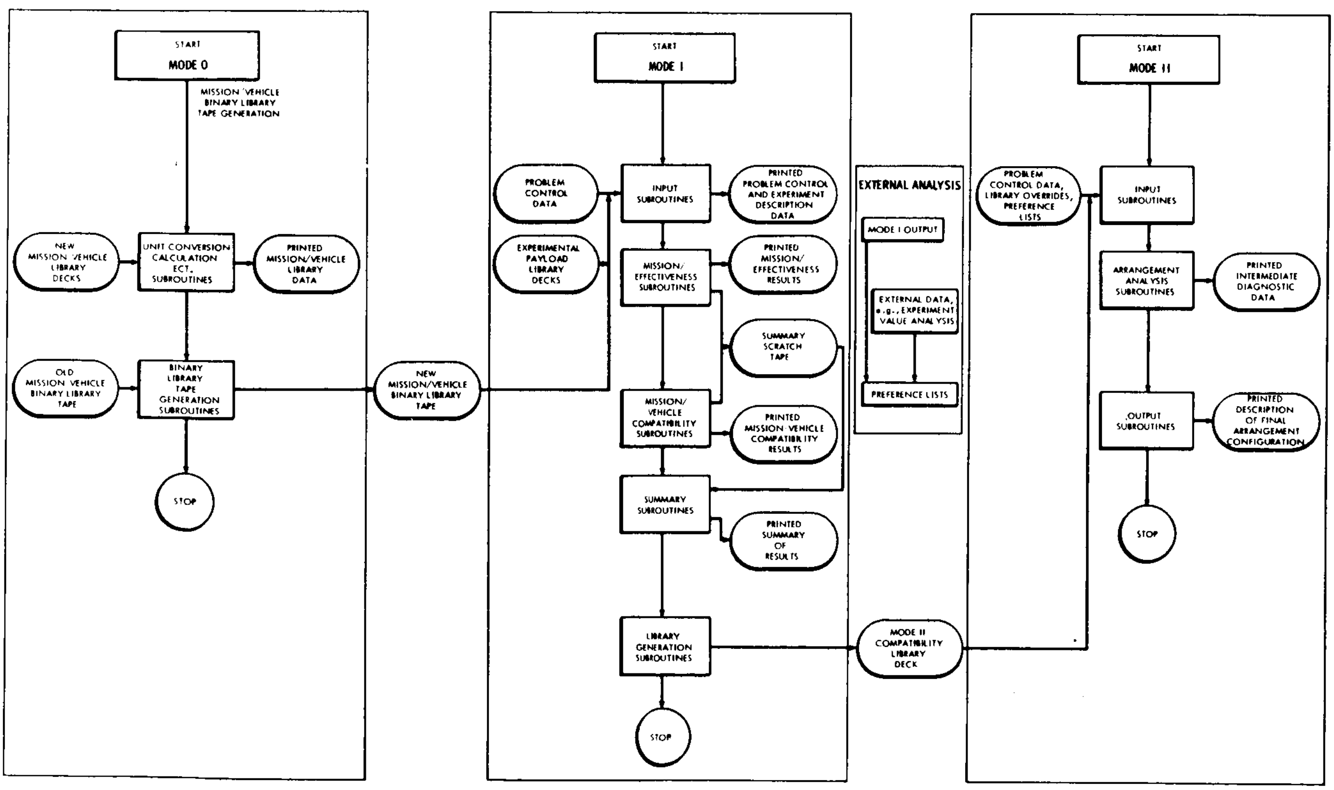


Figure 6-1 SATURN EXPERIMENTAL PAYLOAD TECHNICAL EVALUATION AND RATING

The initial operation of SEPTER must be performed by Mode 0. This mode is not an analysis operation; its function is to perform unit conversions and compile binary library tapes of mission/vehicle/ primary payload characteristics from card decks for direct input to Mode I. The use of a binary library tape makes it possible to decrease computer running time and makes data conversions and the storage of internal and external data other than card decks more efficient.

Mode I is the operational mode in which the compatibility and and effectiveness of single experimental payloads are analyzed. As shown in Figure 6-1, inputs to Mode I consist of (1) mission/vehicle/ primary payload characteristics library data (binary library tape), (2) the experimental payload characteristics library data (card decks), and (3) problem control data (card decks). Problem control data are used to identify tapes, select computational options, and specify overrides for the binary library tape input data. Depending upon the options selected, a limited number or all of the analysis subroutines shown in Figure 6-1 are utilized. For example, either the compatibility or the effectiveness analysis may be selected independently or both analyses may be specified. Similarly, the optional library overrides in the problem control may be used to specify, for example, a new launch date or a different excess payload capability.

The output of Mode I is in the form of printed results and, if specified and applicable, a library deck containing the required input for Mode II. This library deck consists of library data utilized in Mode I plus the computed compatibility results pertaining to individual experiments.

The external analysis required between the operation of Modes I and II consists of the formulation of preference lists to establish the desired order (priority) in which experimental payloads are to be loaded aboard the vehicle for a given mission. Although the effectiveness and compatibility output data of Mode I are obviously useful in the formulation of preference lists, additional data or different methods of establishing priority may be used in arriving at a preference list. Several sets of preference lists may be formulated for a given set of experimental payloads.

Mode II is the operational mode of Program SEPTER which is used to analyze multiple experimental payload compatibility and arrangement configurations aboard the vehicle. The inputs to Mode II consist of a preference list, an experimental payload-mission/vehicle compatibility library deck (from Mode I output), problem data (options and controls), and optional library overrides. The problem data consist of, for example, print-out and placement policy options and controls on the placement policy iteration and cutoff. Optional library overrides consist of the specification of a different excess payload capability and Mode I compatibility overrides (including predetermined placements for arbitrary experiments).

The output of Mode II is in the form of printed results in which the accepted experimental payloads from the preference list and the cavities within which they have been placed (according to the predetermined and optimal arrangement analyses) are listed.

More detailed descriptions of the computer program operations, input data and libraries, the methodology used in the individual areas of analyses, and the output from each mode of analysis are given in subsections 6.2 and 6.3.

6.2 MODE I OPERATION

Mode I analysis consists of the determination of the operational and physical compatibility and the experiment/mission effectiveness of single experimental payloads in terms of a given vehicle/mission/primary payload combination. Compatibility criteria include only major items which are considered significant within the scope of the program philosophy. Experiment/mission effectiveness is computed as the percent of data acquisition objectives accomplished. Effectiveness relationships are determined externally in terms of the initial values of the orbital elements and mission parameters in which the experimental payload is deployed. In Mode I output, the overall GO/NO-GO compatibility, the degree of compatibility or incompatibility, the experiment effectiveness data, and the experimental payload library data for input to Mode II of the program are defined.

6.2.1 Libraries

The mission/vehicle/primary payload data and the experimental payload data for SEPTER are stored in the form of libraries. This method of storage provides a high degree of flexibility in the use of the program. Various combinations of mission/vehicle/primary payload and experimental payloads may be selected at the user's discretion for use in SEPTER. Preliminary definitions may be readily updated. The libraries may be easily modified and expanded to include the missions and payloads of other spacecraft (e.g., Apollo Applications - LEM Lab, NASA Can, etc.).

Two distinct library types are used in Program SEPTER to store and provide definition-type input data:

1. Mission/Vehicle/Primary Payload Characteristics Library

2. Experimental Payload Characteristics Library.

The Mission/Vehicle/Primary Payload Characteristics Library contains the listed definition data:

1. Launch date and time (year, month, day, and solar time at the launch longitude)

2. Launch trajectory parameters (position and velocity as a function of time from launch)

3. Mission duration of primary payload (days)

4. Primary payload separation time (seconds after launch)

5. Vehicle/Mission identification (e.g., SA-207)

6. Vehicle dependent environmental data (electromagnetic signal and sensitivity levels and bandwidths)

7. Vehicle-zone dependent environmental data (acoustics/vibration ambient levels)

8. Vehicle excess payload capability

9. Identification of vehicle experimental payload locations (cavities) - zone number and dash number for each cavity, e.g., 4-3

10. Cavity thermal environmental data (time-space averaged temperature, allowable rate of heat dissipation, and total short period heat dissipation for prelaunch, launch, and orbit mission phases)

11. Cavity structural mass limits

12. Cavity groups structural limits for vehicle zones

13. Cavity volumes

14. Cavity capacities for geometric standard shapes (spheres, cylinders, and rectangular parallelepipeds)

15. Cavity allowable deployment modes.

The Experimental Payload Characteristics Library contains the following definition data (as applicable) for each experiment:

1. Identification (abbreviated name and number)

2. Availability date (year, month, day)

3. Installation time required (days)

4. Deployment mode required (e.g., propulsive separation)

5. Deployment time required (applicable to ejected experimental payloads only)

6. Deployment impulsive velocity increments and angles (applicable to propulsively ejected experimental payloads only)

7. Environmental data (thermal, acoustics/vibration, and electromagnetic)

8. Standard shapes and dimensions (applicable to entire experimental payload for fixed design and to critical component for amorphous design)

9. Standard shape alignment with vehicle axes

10. Total mass

11. Total volume

12. Reliability data

13. Development time (months)

14. Cost data

15. Effectiveness data.

6.2.2 Problem Input and Controls

The inputs for the operation of Mode I consist of library data and problem control data. The Mission/Vehicle/Primary Payload Characteristics Library data are provided from a binary library tape. The Experimental Payload Characteristics Library data are provided from card decks. Both types of libraries are required to run a problem. Problem control data (from card decks) are used to select computational options and specify optional overrides for the binary library tape input data.

Problem control options and optional library overrides provide operation and program utilization versatility. Problem control options include the following:

1. Computation of experiment/mission effectiveness and experimental payload-mission/vehicle compatibility for each experimental payload. Compatibility is computed with respect to each vehicle cavity as well as to the overall vehicle.

2. Computation of experiment/mission effectiveness data only

3. Computation of experimental payload-mission/vehicle compatibility data only

4. Generation of a Mode II compatibility library card deck

5. Selection of experimental payload cavities (specify how many and which ones).

Mission/Vehicle/Primary Payload Library data overrides include the following:

1. Launch date

2. Launch time

3. Vehicle-primary payload separation time (allowing ejection of experimental payloads)

4. Primary mission duration

5. Excess payload capability.

6.2.3 Deployment Methodology

The various data acquisition objectives of candidate experimental payloads require a computer methodology which is able to simulate the deployment of each experimental payload into its required operating environment at the proper time in the mission. For example, some experimental payloads may be able to acquire most or all of their data objectives by simply remaining fixed inside the launch vehicle. Other experiments may require extension of an antenna or some piece of equipment during their operation, whereas another type of experimental payloads may require separation from the vehicle or injection into an orbit different from that of the primary payload.

The deployment methods used in the computer program to simulate the placement of experimental payloads into their required operating environments are called deployment modes. As defined for the program, a deployment mode is not limited to actual ejection of the experimental payload from the vehicle.

The deployment methodology of the computer program consists of

1. A defined set of deployment modes which adequately cover the operating requirements of candidate experimental payloads

2. The coordinate systems in which the deployment modes are defined

3. The program logic used to calculate the orbital elements and mission parameters of an experimental payload for any specified deployment mode and deployment time.

6.2.3.1 Deployment Modes

The six deployment modes selected for the computer methodology of Program SEPTER are listed below with their identification number and their defining characteristics.

Mode 0 - Fixed

1. The experimental payload remains on the vehicle throughout the mission.

2. The extension of only an antenna is required.

3. No exposure to a vacuum is required.

Mode 1 - Fixed Exposed

1. The experimental payload remains on the vehicle throughout the mission.

2. The extension of only an antenna is required.

3. Exposure to a vacuum is required.

Mode 2 - Extension

1. The experimental payload remains on the vehicle throughout the mission.

2. The extension of components other than an antenna is required.

Mode 3 - Separation

1. The experimental payload is separated from the vehicle. The orbital elements of the separated payload are assumed to be the same as those of the vehicle at the time of deployment. (Subsequently, the orbital elements of the vehicle and payload may differ because of perturbative forces, e.g., atmospheric drag).

2. No propulsion is required.

Mode 4 - Propulsive Separation (ΔV)

1. The experimental payload is separated from the vehicle.

2. Propulsion is required to inject the experimental payload into an orbit different from that of the primary payload.

Mode 5 - Recovery Capsule Separation

1. The experimental payload remains on the vehicle throughout the mission.

2. Separation of one or several data recovery capsules is required.

These deployment modes are pictorially illustrated in Figure 6-2.

6.2.3.2 Coordinate Systems

The coordinate systems used in the deployment methodology of the program are shown in Figure 6-3. The position of the vehicle or experimental payload is specified in terms of altitude, longitude, and latitude with respect to a spherical, rotating earth. Velocity is defined by inertial speed, flight path angle, and azimuth angle.

The coordinate system in which the ΔV is defined for propulsive separation (Mode 4) has an axis tangential to the velocity vector at the time of deployment, an axis normal to the velocity vector in the orbit plane of the vehicle, and a lateral axis normal to the orbit plane of the vehicle. The in-plane thrust (ΔV) angle, θ, is measured from the velocity direction to the projection of the ΔV vector on the orbit plane. The out-of-plane angle, ϕ, is measured from the orbit plane to ΔV vector.

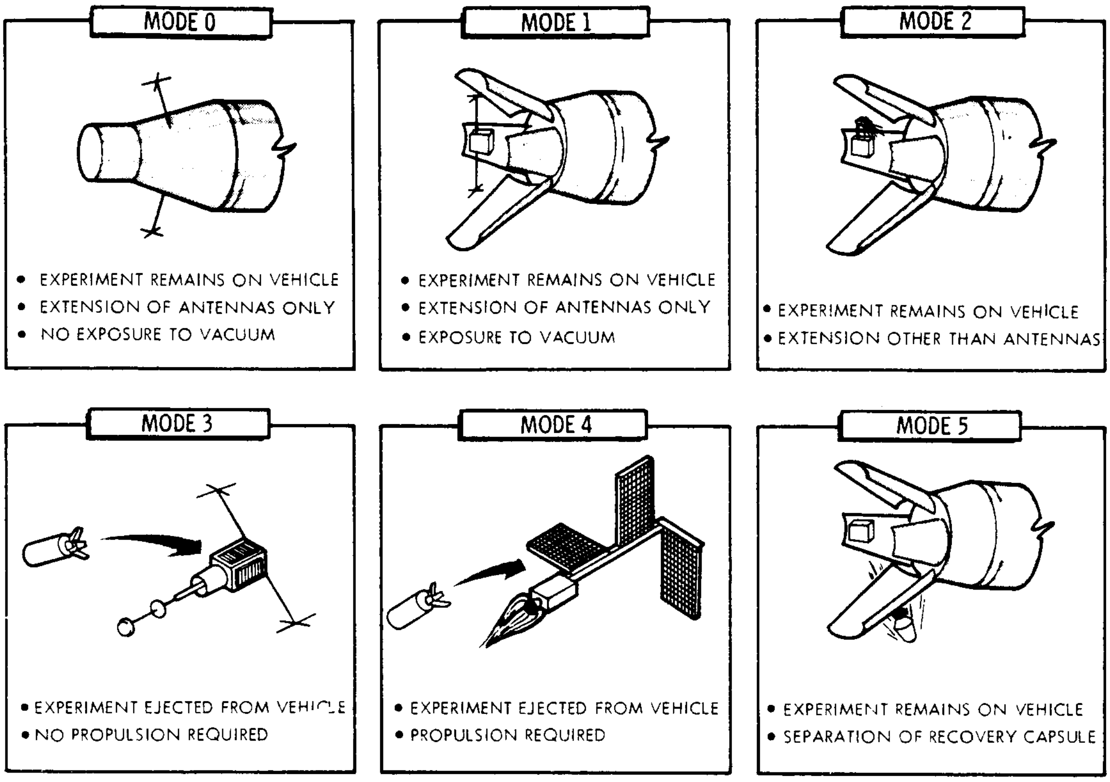


Figure 6-2 OPERATIONAL REQUIREMENTS

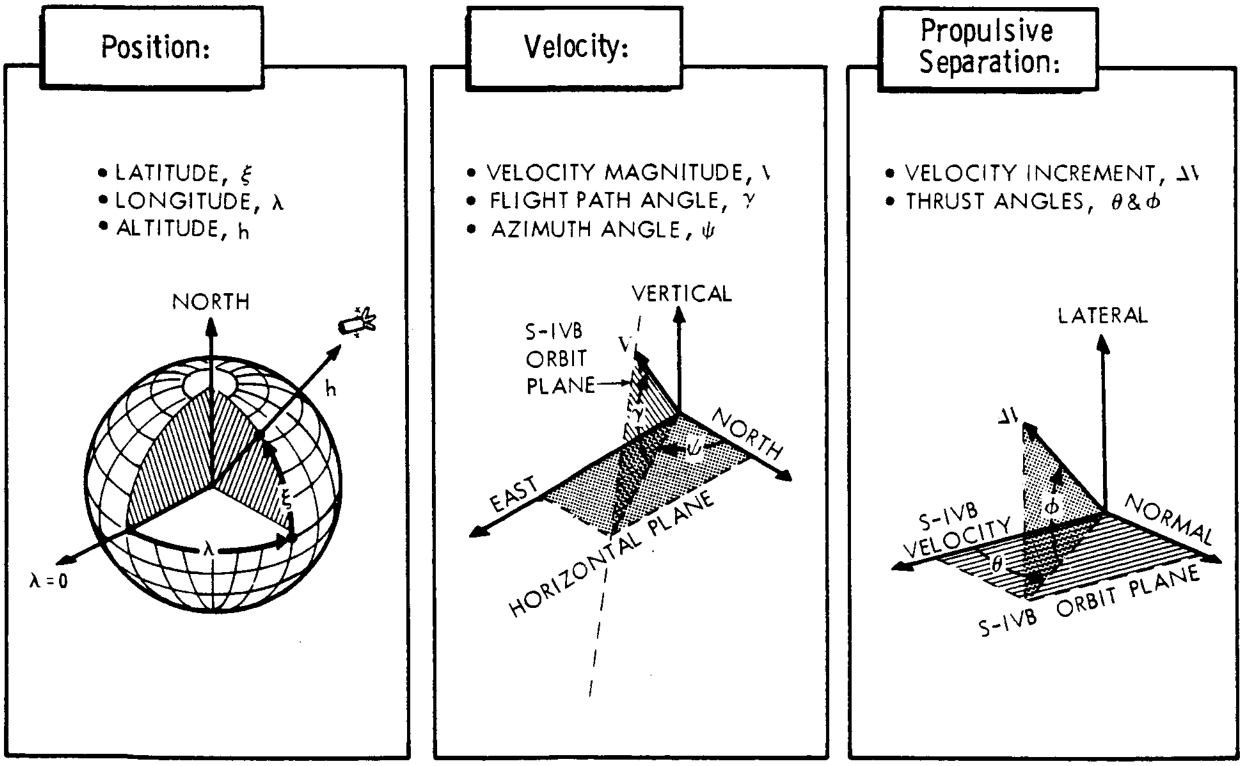


Figure 6-3 DEPLOYMENT METHODOLOGY COORDINATE SYSTEMS

6.2.3.3 Orbital Elements and Mission Parameters

The methodology used in the computer program to calculate the orbital elements and mission parameters of an experimental payload for any specified deployment mode and deployment time is illustrated in Figure 6-4.

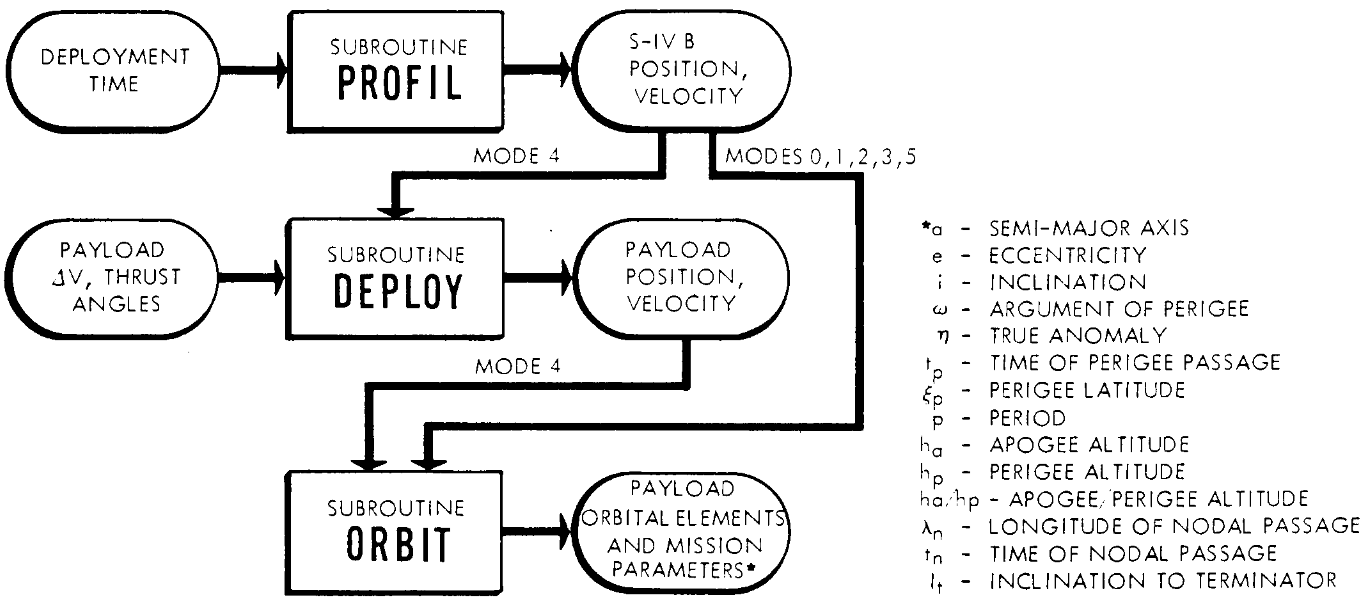


Figure 6-4 ORBITAL ELEMENTS LOGIC

Subroutines PROFIL, DEPLOY, and ORBIT are used to calculate the mission profile parameters and orbital elements at deployment time. Initially, the position and velocity of the vehicle are computed at deployment time in subroutine PROFIL. These data are obtained from the Mission/Vehicle/Primary Payload Characteristics Library. When deployment Modes 0, 1, 2, 3, and 5 are specified, the position and velocity data obtained from PROFIL are used in subroutine ORBIT to compute the initial orbital elements of the experimental payload. When Mode 4 is specified, the velocity vector obtained from PROFIL is modified by the use of subroutine DEPLOY before calculation of the initial orbital elements in subroutine ORBIT.

The orbital elements and additional mission parameters calculated for any deployment mode and deployment time are listed below. These data are made available to another subroutine (EFFECT) in the program to calculate experiment/mission effectiveness.

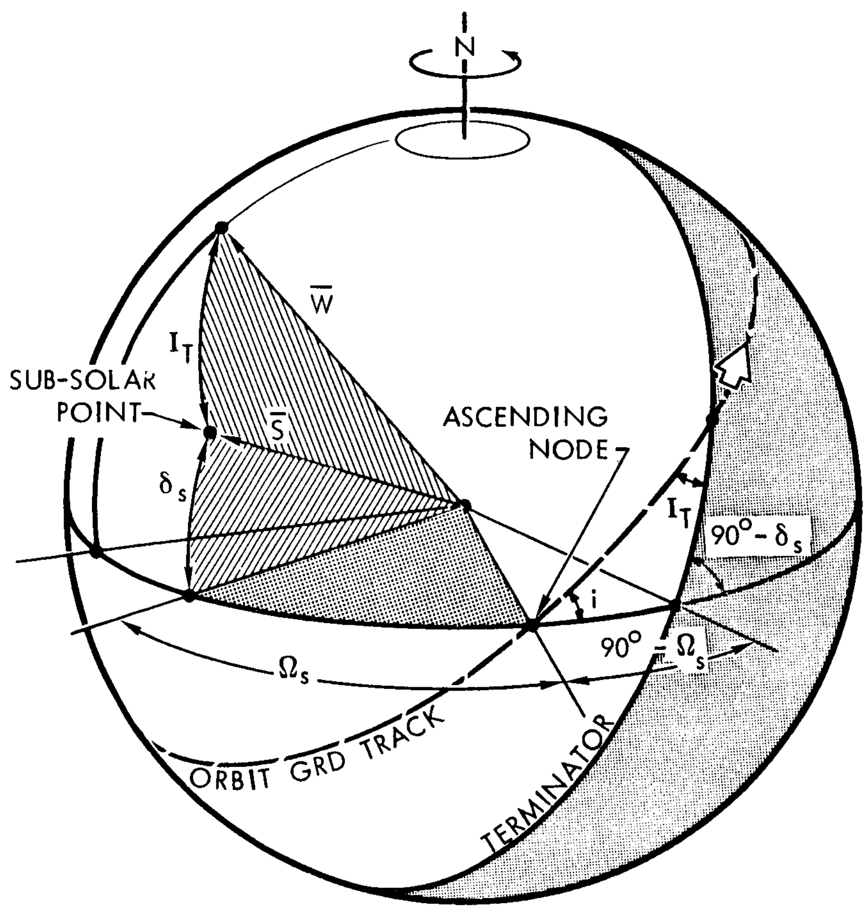
Orbital Elements and Mission Parameters

1.	Semi-major axes	a
2.	Eccentricity	e
3.	Inclination	i
4.	Argument of perigee	ω
5.	True anomaly	η
6.	Time pf perigee passage	t_p
7.	Perigee latitude	ξ_p
8.	Orbital period	P
9.	Apogee altitude	h_a
10.	Perigee altitude	h_p
11.	Apogee/perigee altitude	h_a/h_p
12.	Longitude at nodal passage	λ_n
13.	Time of nodal passage	t_n
14.	Inclination to terminator	I_T
15.	Solar declination	δ_s

Since solar declination is not a function of the deployed orbit for an experimental payload, its value is computed in the main program of SEPTER as a function of launch date. The solar declination angle and inclination to the terminator angle are measured as illustrated in Figure 6-5.

6.2.4 EXPERIMENT/MISSION EFFECTIVENESS METHODOLOGY

The relationships between experiment effectiveness factors and the initial orbital elements and/or mission parameters, listed in Table 5-1, are established during the external experiment effectiveness



I_T = inclination of orbital plane to plane of terminator

i = inclination of orbital plane to equatorial plane

$\overline{W}$ = orbit angular momentum unit vector

$\overline{S}$ = solar position unit vector

δ_s = solar declination

Ω_s = longitudinal displacement between ascending node and sub-solar point

Figure 6-5 ORBIT/TERMINATOR GEOMETRY

analysis discussed in Section 5. The effectiveness functions are loaded into the effectiveness segment of the Experimental Payload Characteristics Library in tables of one- and/or two-dimensional arrays. Effectiveness factors are obtained from each table by use of table "look up" procedures and are multiplied together to obtain the absolute value of experiment effectiveness (percent accomplishment of data acquisition objectives). The effectiveness tables are entered with the computed values of one (or two) of the independent variables listed in Table 5-1.

The percent effectiveness relative to the maximum possible effectiveness (absolute effectiveness divided by maximum effectiveness), designated "normalized effectiveness", is also computed. In general, the maximum effectiveness possible for a given experimental payload can be less than 100 percent. The maximum possible effectiveness value is an input quantity which corresponds to the effectiveness of the design orbit. (See Section 5.1.3).

Table look-up procedures are provided for the three types of effectiveness factor relationships shown in Figure 6-6: (1) effectiveness factor as a continuous function of two variables, (2) effectiveness factor as a step function of two variables, and (3) effectiveness factor as a function of one variable (step or continuous). Options are provided for either linear or fourth-order Lagrange interpolation.

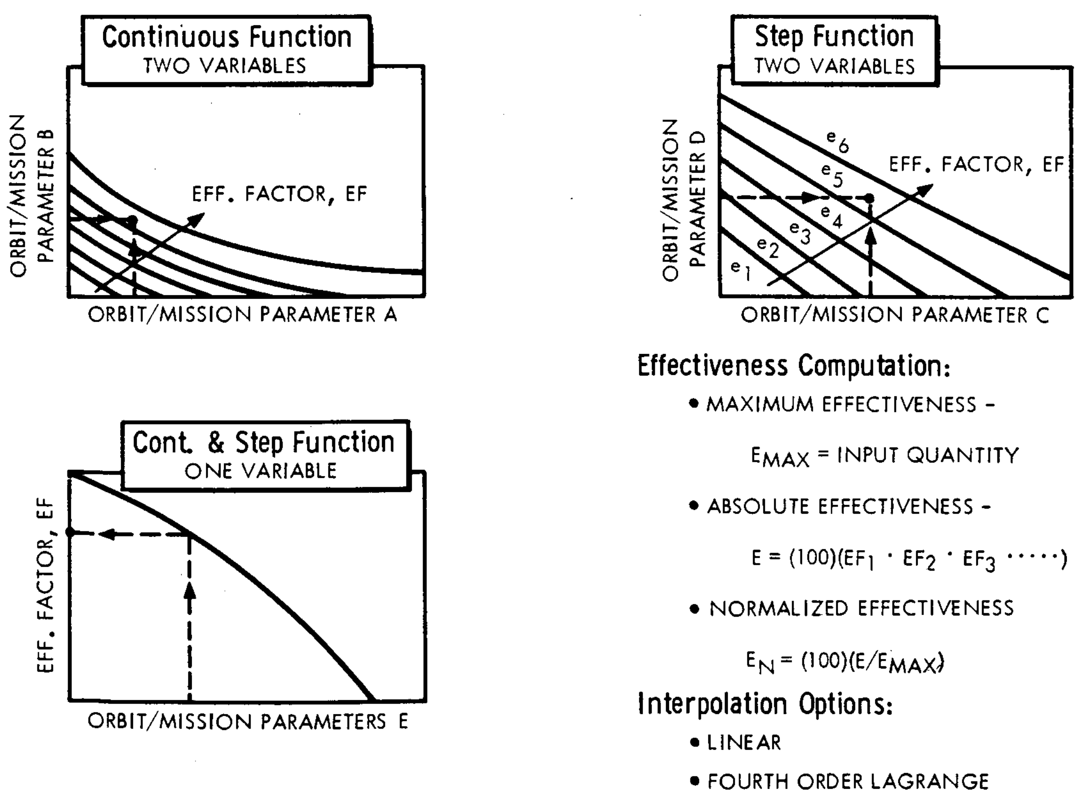


Figure 6-6 COMPUTATION OF EXPERIMENT/MISSION EFFECTIVENESS

6.2.5 Experimental Payload - Mission/Vehicle Compatibility Methodology

The computer methodology used to determine experimental payload compatibility with both the primary payload mission and the vehicle is based on two fundamental guidelines:

1. The experimental payload must tolerate all mission and vehicle constraints and environments.

2. The experimental payload cannot significantly affect the primary mission/vehicle performance.

The compatibility criteria and the methodology used to determine compatibility are described in the subsections 6.2.5.1 through 6.2.5.4.

6.2.5.1 Compatibility Criteria

The criteria used in the computer program to determine experimental payload-mission/vehicle compatibility are classified as (1) operational and (2) physical. The specific criteria in each category are as follows:

Operational Criteria

1. Experimental payload availability

   a. Availability date

   b. Launch date

   c. Installation time required

2. Experimental payload deployment

   a. Mode

   b. Time (if applicable)

Physical Criteria

1. Environmental

   a. Thermal

   b. Acoustics

   c. Vibration

   d. Electromagnetic

2. Mass attachment limit

3. Volume/geometry

   a. Size

   b. Shape

   c. Orientation.

In order for an experimental payload to be mission/vehicle compatible, it must satisfy the GO condition for the following list of criteria:

1. Thermal
2. Acoustic
3. Vibration
4. Mass attachment
5. Volume/geometry
6. Deployment mode
7. Deployment time (if applicable).

The remaining criteria, i.e., experimental payload availability and electromagnetic interference (EMI) are included to indicate possible problem areas which should be investigated more thoroughly by external analyses.

The acoustic and vibration criteria are treated uniquely in the program in that a NO-GO condition is initially corrected to a GO condition by the computation of a mass penalty. In the event that the computed mass penalties are obviously excessive, the NO-GO condition remains to affect the overall compatibility decision for the experimental payload. The unique aspects of all compatibility criteria and how they are used in the program methodology are given in subsections 6.2.5.2 through 6.2.5.4.

Some additional criteria were considered for use in the computer program, but were eliminated because of their nonapplicability and/or incompatibility with the scope and philosophy of the present computer program. These are primarily environmental criteria, such as:

1. Acceleration/shock

2. Humidity

3. Pressure/vacuum

4. Contamination (dust, fungus, gases, and salt spray)

5. Radiation

In the present study, the assumption was made that experimental payloads would be designed to survive normal launch pad, boost and orbit environments that are not peculiar to a specific vehicle or vehicle locations (i.e., zones or cavities).

In general, the environmental criteria that were eliminated are those that the experimental payloads would normally be designed to survive or that would be relatively constant for all vehicle locations. Compatibility checks for relatively constant criteria can best be accomplished external to the computer program by the assignment of payload qualification specifications compatible with the launch vehicle.

The radiation environmental criterion may be vehicle-location dependent in a limited number of cases. However, it was considered a low priority item which did not warrant the apparent extensive investigation required for a meaningful compatibility methodology in the present computer program.

6.2.5.2 Experimental Payload Availability Date/Launch Date Compatibility

The experimental payload must obviously be available for installation in the launch vehicle prior to the launch date to be compatible. However, this simple check does not allow for the time required for installation and/or checkout of the experimental payload. Therefore, for a more realistic compatibility check, an installation time can also be specified. Thus, in the compatibility check, the experimental payload availability date plus the required installation time (days) must precede the launch date.

The availability date/launch date compatibility determination in the program does not affect the overall experimental payload/mission/vehicle compatibility decision. If a GO-condition is calculated, the number of "buffer" days are given in the output. In the case of a NO-GO calculation, a warning statement is given in the output.

6.2.5.3 Deployment Compatibility

Deployment compatibility of an experimental payload with a vehicle cavity is based on two criteria: (1) deployment mode and (2) deployment time (if applicable). These are operational compatibility criteria. The overall GO or NO-GO compatibility determination in the computer program is affected by both of these criteria.

Each cavity is assigned the deployment mode(s) which it can accommodate. These modes are defined in subsection 6.2.3.1. Likewise, each experimental payload is assigned its required deployment mode. The assigned mode data are stored in the Mission/Vehicle/Primary Payload Characteristics Library and the Experimental Payload Characteristics Library, respectively. Deployment mode compatibility is simply a check of the required mode with the available mode for the cavity from which the experimental payload must be deployed.

Deployment time compatibility is dependent upon the assigned times at which a cavity is available for the specified deployment mode and the assigned time at which an experimental payload must be deployed in the mission. Only deployment modes 3, 4, and 5 (requiring separation or ejection of an experimental payload from the vehicle) are time dependent.

Deployment direction compatibility was considered for incorporation in the computer program. However, currently available data on the attitude control system of the Saturn IU/SIVB (as given in References 6-1, 6-2, and 6-3) indicate that for post-separation operations, the attitude control system is capable of maintaining almost any commanded orientation. For the basic configuration used for the study (Saturn IB/Apollo vehicle/payload), experimental payloads cannot be ejected from the vehicle until after (1) separation of the spacecraft from the vehicle, (2) deployment of the LEM adapter panels, and (3) removal of the LEM from the adapter. Therefore, the assumption was logically made that any required deployment direction can be achieved for deployment modes 3, 4, and 5 without violating mission/vehicle constraints and without affecting primary mission/vehicle performance.

6.2.5.4 Environmental Compatibility

Environmental compatibility is based on criteria which are either cavity dependent, vehicle zone dependent, complete-vehicle dependent, and, in some cases, mission phase dependent. The environmental criteria used in the computer program are (1) thermal, (2) acoustics/vibration, and (3) electromagnetic. The compatibility methodology used in the computer program for each environmental criterion is discussed in detail in the following subsections.

6.2.5.4.1 Thermal. Thermal environment compatibility is determined by comparing thermal parameter values pertaining to a specific cavity or vehicle zone with the corresponding parameter values associated with a given experimental payload. This comparison of values will yield GO or NO-GO decisions. Thermal compatibility requires a full set of GO decisions for the thermal parameters that are selected as meaningful for a given cavity-experimental payload.

The three following thermal parameters are used for comparison in the program. Two of these parameters, items 2 and 3, are optional.

1. Time-space averaged sink temperature

2. Heat dissipation rate

3. Total short period heat dissipation

These parameters are defined for three mission phases: (1) prelaunch, (2) launch, and (3) orbital. Although some of the experimental payloads are ejected from the vehicle and must, therefore, also be compatible with the orbital operational environment, this environment is not a function of the spacecraft and must be considered in experiment design. Compatibility checks are made only during the mission phases where the experimental payload is aboard the vehicle.

In certain situations, the heat dissipation rate and the total short period heat dissipation are not mutually exclusive criteria. For example, suppose the heat dissipation rate exceeds the allowable rate for a particular cavity for a short duration experiment. If the length of time is short, the total short period heat dissipation may be well below the tolerable level. In this case, the total short period heat dissipation may be a more meaningful basis of comparison. Conversely, for some experiments the heat dissipation rate may be the meaningful parameter for comparison. Therefore, the computer program methodology provides an optional capability such that either the heat dissipation rate or the total short period heat dissipation can be excluded from the compatibility checks. However, this optional capability does not preclude the use of both parameters in the compatibility checks in cases where they are both applicable. The optional control is provided for each experimental payload in the Experimental Payload Characteristics Library.

The computer program methodology for thermal compatibility checks is illustrated for the GO condition in Figure 6-7.

Guideline:

- EXPERIMENT MUST TOLERATE CAVITY THERMAL ENVIRONMENT WITHOUT AFFECTING THE PRIMARY PAYLOAD AND VEHICLE

Approach:

- DEFINE CAVITY THERMAL ENVIRONMENT FOR EACH MISSION PHASE (e.g., PRE-LAUNCH, LAUNCH, ORBIT)
- DEFINE LIMITS OF ACCEPTANCE FOR EXPERIMENT PAYLOAD THERMAL PARAMETERS
- DEFINE ANALYTICAL METHODOLOGY FOR THE DETERMINATION OF THERMAL COMPATIBILITY

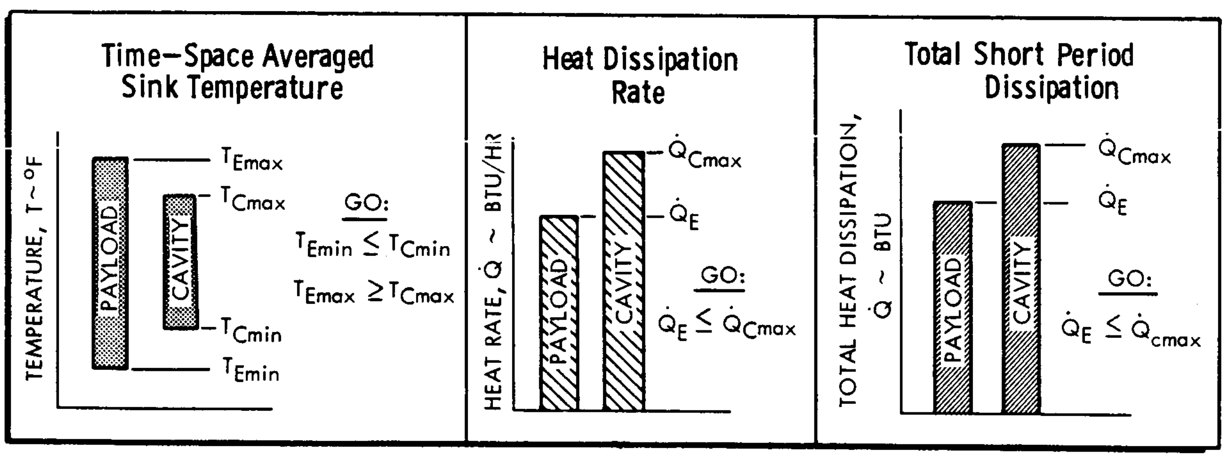


Figure 6-7 THERMAL ENVIRONMENT COMPATIBILITY

6.2.5.4.2 Acoustics and Vibration. Acoustics and vibration environmental compatibility determination in the computer program does not yield a direct GO or NO-GO decision. Rather, the compatibility methodology determines whether a given experimental payload can survive the environmental noise level and vibration level induced during booster operation. If a possible failure is indicated, a mass penalty is computed and added to the mass of the experimental payload in order to correct the tolerance deficiency. The experimental payload is disqualified (NO-GO) only if the calculated mass penalty is obviously excessive.

The direct GO or NO-GO decision methodology is not utilized for the acoustics/vibration compatibility determinations for reasons that are apparent after investigation of the unique aspects associated with these environmental criteria.

In the simple GO or NO-GO comparison concept, each cavity (or vehicle zone) available for an experimental payload is assigned ambient noise and vibration levels induced by the operating booster.

Also, each experimental payload is assigned noise and vibration tolerance levels. If the tolerance levels are equal to or greater than the ambient levels of the cavity (or zone) being considered, the experimental payload is compatible with the vehicle cavity or zone. Conversely, if the tolerance levels fall below the ambient levels, the experimental payload is susceptible to failure. For a given "fixed-design" experimental payload, this method of analysis approaches the actual situation and would be adequate for the computer methodology. However, since the experimental payloads defined for Program SEPTER are not necessarily "fixed-design," preliminary data will seldom be accurate enough to justify the elimination of an experimental payload on this basis. Also, even if the actual tolerance levels fall below the ambient levels, it is reasonable to assume that the experimental payload components could be built to withstand the environmental levels by the addition of increased material gauges, isolation, mountings, stiffeners or dampers. However, these deficiency correction methods would incur mass penalties.

A meaningful compatibility methodology used in Program SEPTER consists of the assignment of ambient noise and vibration levels and tolerance noise and vibration levels as stated for the simple GO or NO-GO comparison concept. However, since it was found reasonable to assume mass penalties for the correction of deficiencies in a "non-fixed-design" experimental payload, this feature is also a part of the methodology. Mass penalties are calculated for both noise and vibration deficiencies. A NO-GO compatibility is only given if the calculated mass penalties are obviously excessive when compared with the payload mass. The limit mass penalty has been arbitrarily set equal to the mass of the acoustic/vibration susceptible components of the experimental payload.

The noise and vibration parameters and the methodology used in the compatibility checks are illustrated in Figure 6-8. The ambient levels of these parameters are given for vehicle zones. The corresponding tolerance levels are given for experimental payloads. The mass penalty methodology used to correct a deficiency (NO-GO condition) to a GO condition is illustrated by the data and equations given in the lower half of Figure 6-8.

The compatibility methodology used for acoustics/vibration environmental criteria is based on the following assumptions:

1. If an experimental payload can survive the launch phase of a mission it can survive all other phases.

2. The experimental payload does not operate during the launch phase.

3. Acoustic and vibration tolerance deficiencies can be corrected by the addition of mass (in the form of heavier material, isolation mountings, stiffeners, or dampers) to the experimental payload.

4. The mass required to correct a deficiency is proportional to the payload density, the percent deficiency, and the mass of susceptible components.

5. The correction of an acoustic tolerance deficiency does not correct a vibration tolerance deficiency and vice versa. While the correction mass penalties that improve vibration tolerance generally also improve noise tolerance, the range of frequencies associated with each are different. Therefore, mass used to solve a vibration susceptibility problem in a given component will not necessarily solve its acoustic susceptibility problem.

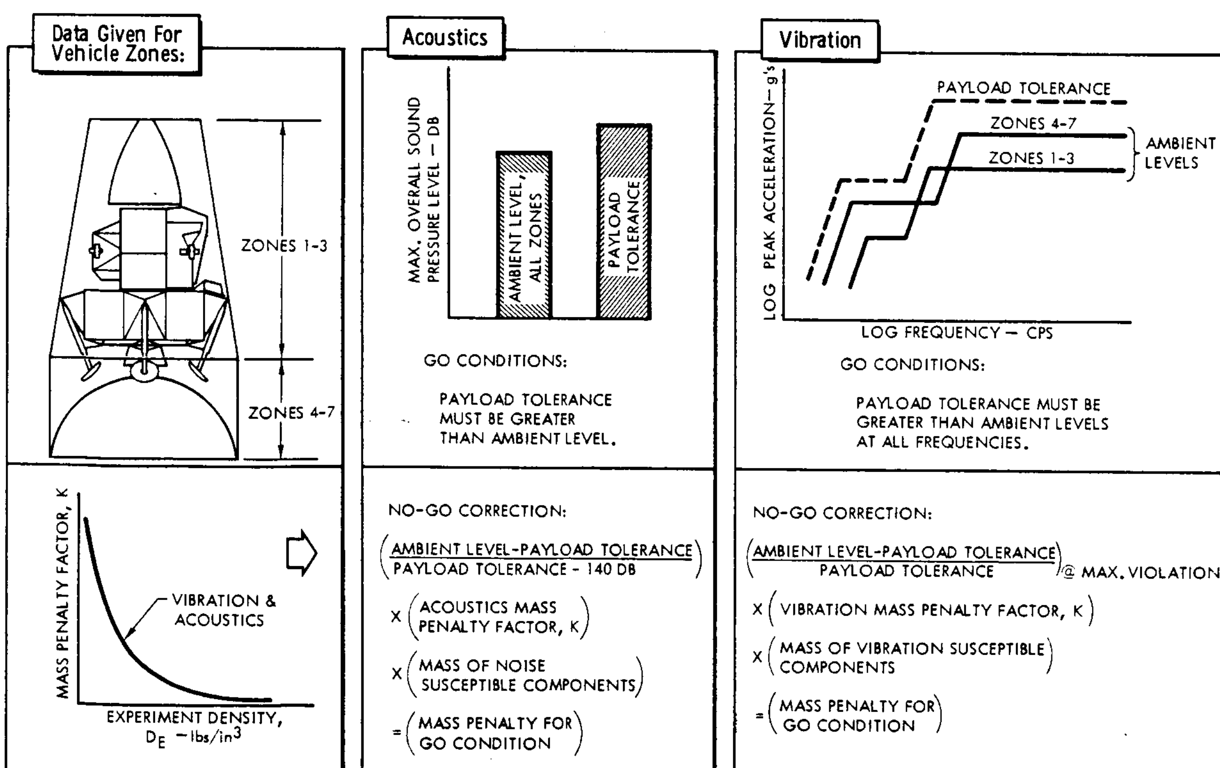


Figure 6-8 ACOUSTICS AND VIBRATION ENVIRONMENT COMPATIBILITY

6.2.5.4.3 Electromagnetic. Electromagnetic compatibility can be defined as the ability of each piece of electrical/electronic equipment in an integrated system to perform its design function without interfering with the performance of the designed function of any other piece of electrical/electronic equipment in the system. In the present study, the Saturn vehicle/primary payload-experimental payload can be regarded as the system.

The basic parameters which are used to determine if one equipment will interfere with the function of another are

1. Level and bandwidth of the signal the equipment is capable of emitting.

2. Level and bandwidth of the signal to which the equipment is capable of responding

3. "Coincident time interval" or the occurrence of simultaneous operation of equipment in which the parameters, items 1 and 2, overlap

4. Amount of isolation (insulation or separation) between equipment.

In order to determine compatibility using the given parameters, the following sequence of checks must be made:

1. Compare signal levels of "emitters" with sensitivity levels of receivers. If they do not overlap, compatibility exists. If they do overlap, continue with sequence step 2.

2. Compare bandwidths of "emitters" with bandwidths of receivers. If they do not overlap, compatibility exists. If they do overlap, continue with sequence step 3.

3. Determine if the equipment which had overlapping parameters in both steps 1 and 2 are operated simultaneously. They are compatible if they do not operate simultaneously. If they do operate simultaneously, they are incompatible and sequence step 4 must be checked.

4. Determine the amount of isolation between equipment of step 3. If the isolation is equal to or greater than the overlap given in step 1, the equipment compared are compatible; otherwise, an incompatibility exists.

If the given methodology were accurately defined and applied in the computer program, a definite GO or NO-GO decision could be determined. However, the use of this "exact" methodology is considered not within the scope of the present study and computer program because (1) sequence step 3 requires vehicle/primary payload/experimental payload operations scheduling and (2) the determination of the amount of isolation between equipment, required in sequence step 4, is a function of many undefined variables. Also, definition-type data presently available for the experimental payloads and vehicle electromagnetic equipments are not sufficiently accurate.

A simplified methodology has been incorporated in the computer program to determine electromagnetic compatibility. However, because of the simplifying assumptions made, the compatibility checks are not sufficiently accurate or complete to yield a definite GO or NO-GO decision. Therefore, the output of the computer program only gives warning-type statements for indicated incompatibilities. Frequency ranges are given where incompatibilities may exist. The output is helpful in locating possible problem areas which can only be thoroughly analyzed external to the computer program.

The simplifying assumptions made in the computer program methodology are as follows:

1. All electrical/electronic equipment operate simultaneously. (The present computer program does not include operations scheduling).

2. No isolation exists between equipments aboard the vehicle.

3. An infinite amount of isolation exists between the vehicle equipment and an ejected experimental payload equipment.

The computer methodology used to check electromagnetic compatibility for both narrowband and broadband types of equipment is illustrated in Figure 6-9. Note that the amount of "overlap", i.e., signal level greater than sensitivity level, is the amount of interference within a given frequency bandwidth and that in order for compatibility to exist, isolation equivalent to the overlap must be provided.

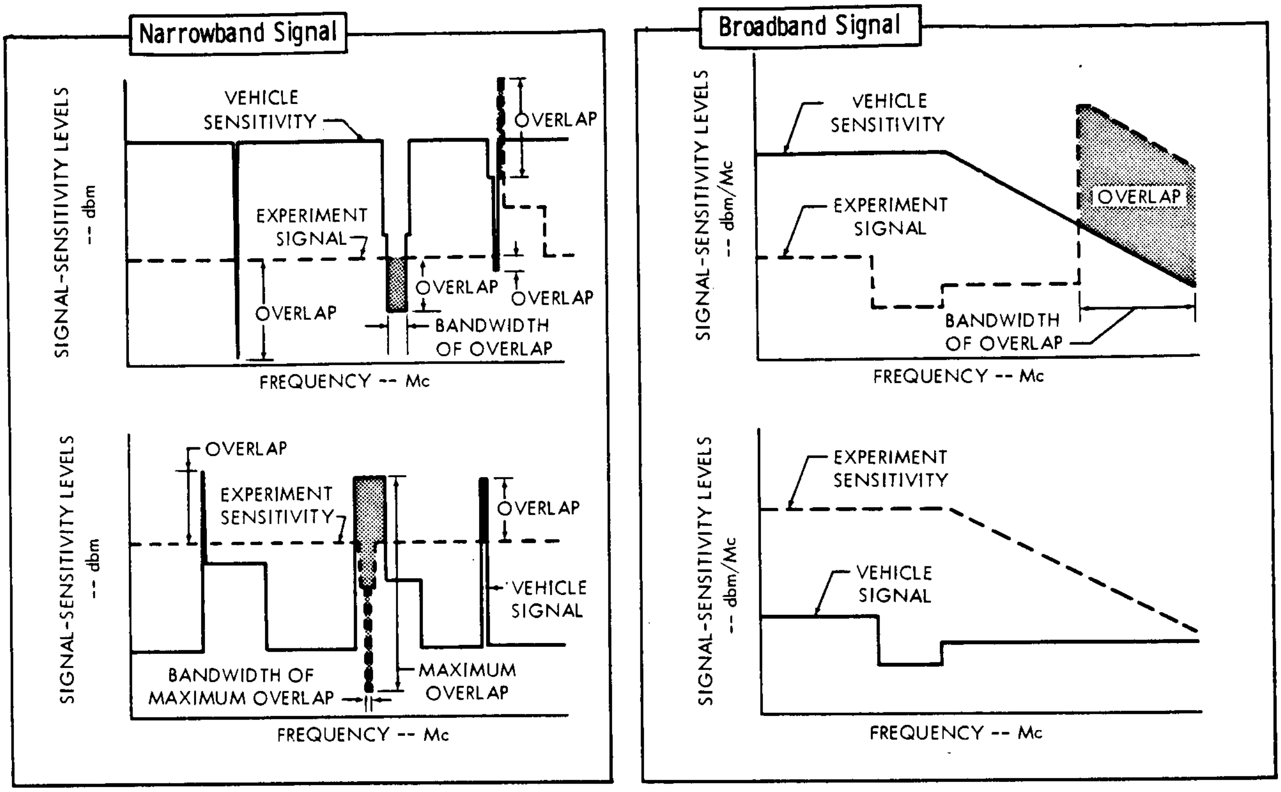


Figure 6-9 ELECTROMAGNETIC ENVIRONMENT COMPATIBILITY

6.2.5.4.4 Mass Attachment. The determination of the mass compatibility of an experimental payload with the cavity in which it is placed is based on the mass attachment limit of the cavity. This limit is usually determined from structural analyses. The total mass of an experimental payload includes any penalty masses which may have been computed and added for the correction of acoustical/vibration tolerance deficiencies.

The experimental payload-cavity mass attachment compatibility methodology in the computer program yields a GO/NO-GO decision for each cavity. However, the methodology which determines the compatibility of an experimental payload mass with the excess payload capability of the vehicle/primary payload does not affect the overall GO/NO-GO decision. In the event that a single experimental payload mass exceeds the total excess payload capability, a warning-type statement indicating the overload is given in the output. In this manner, all other compatibility criteria are analyzed in Mode I, and the final accumulative multiple payload mass compatibility is determined in Mode II.

6.2.5.4.5 Geometric Capacities. The volume/geometry physical compatibility analysis consists in the determination of whether a given experimental payload can be contained within a given payload cavity. The formulation of a general "exact" methodology for the computer program would require an extremely complex computer program logic and in many cases prohibitive data storage capabilities. Therefore, a less general methodology (i.e., one restricted to standard shape representation for experimental payloads) is used. The actual sizes and shapes of the cavities are not represented in the program; instead, their capacities for several geometrical solids are stored in tabular form in the Mission/Vehicle/Primary Payload Characteristics Library. The experimental payloads are represented as either fixed in shape or amorphous. The fixed geometry representations are restricted to one of the standard shapes for which the cavity capacities have been analyzed, i.e., sphere, right circular cylinder, or rectangular parallelepiped. An amorphous geometry payload is treated as a fluid volume containing an undistortable component which is given a fixed geometry representation (standard shape envelope). The total volume of an amorphous geometry payload is composed of the sum of the volumes of the components multiplied by a "packaging" factor. These two concepts allow for the representation of experimental payloads which are (1) in the "off-the-shelf" or final design stages or (2) in the conceptual design stage and amenable to some rearranging of the components of the entire payload package.

The vehicle cavities are divided into two categories, rectilinear or tapered, according to the form of their capacities data. There are very simple methods available for representing the rectilinear capacities in a computer program. However, these methods are not applicable to the tapered cavity capacities. A single technique, which is reasonably simple and nearly exact, was utilized for representing both types of cavities. The method is general, efficient, and accurate.

6.2.5.4.5.1 Sphere and Cylinder Capacities. Since some experimental payloads may require a specified orientation in the vehicle, an orthogonal coordinate system is used in each cavity. In this system, the vertical axis is parallel to the vehicle longitudinal axis, the radial axis emanates from the center of the vehicle and passes through the cavity, and the lateral axis is perpendicular to the other two axes. The geometric capacities compatibility methodology is used to determine whether a sphere of given diameter or a cylinder of given diameter and length with its axes parallel to the vertical, radial, or lateral axes can be contained in a specified cavity.

A tapered and a rectilinear cavity and sphere and cylinder capacities are shown in Figure 6-10. There is only one maximum diameter sphere which can be contained in any cavity and there is obviously no need to specify an orientation. The methodology to determine the geometric compatibility of a spherical experimental payload consists in analytically checking whether the diameter of the payload is less than or equal to the maximum diameter of the sphere which the cavity will contain.

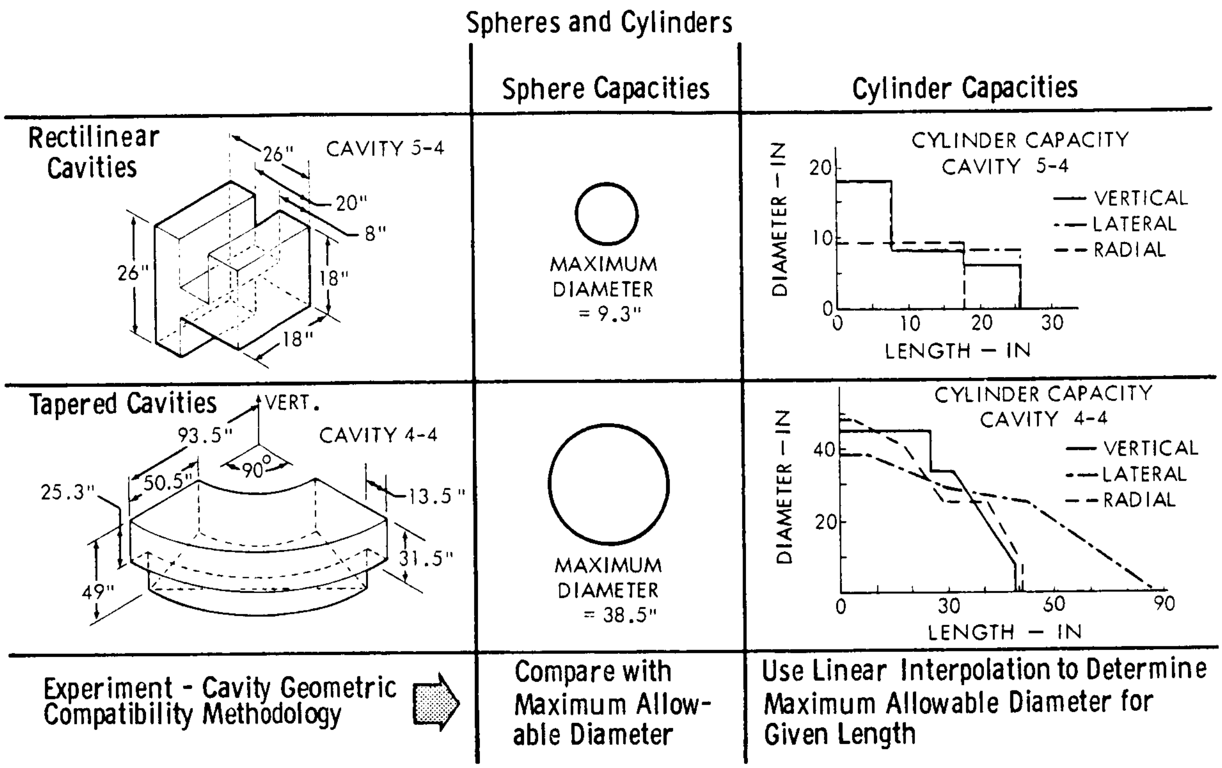


Figure 6-10 GEOMETRIC CAPACITIES COMPATIBILITY 1

The cylinder capacity curves shown for a specified cavity in Figure 6-10 represent the maximum diameters of cylinders of given length and orientation which can be contained in the cavity. A difference between rectilinear and tapered cavities is now evident. The rectilinear-cavity cylinder capacity curves consist of constant diameter steps, and the curves can be replaced by five distinct cylinders, at least one of which will contain any cylinder that can be contained in the cavity. Capacity curves for the tapered-cavity cylinder consist of sloped and curved lines which represent the simplest form in which the cylinder capacities for tapered cavities may be represented.

The methodology used to determine the geometric capacities compatibility of a cylindrical payload consists of the analytical interpolation, from these curves, of the maximum possible diameter corresponding to the length of the payload. The payload diameter must be less than or equal to the maximum possible diameter for compatibility. If the orientation is specified, only one interpolation is required, but if no orientation is specified, as many as three interpolations may be required to determine whether a fit is possible.

6.2.5.4.5.2 Rectangular Parallelepiped Capacities. Because three dimensions are required to specify the size of a rectangular parallelepiped, the capacities for this standard shape payload are represented by surfaces. The rectangular parallelepiped capacity surfaces for the cavities given in Figure 6-10 are illustrated in Figure 6-11. These surfaces represent the maximum possible vertical dimension for each pair of lateral and radial dimensions which can be contained in the cavity. They are shown in Figure 6-11 in both contour and isometric form for clarity. Because each point on a surface corresponds to a rectangular parallelepiped and different orientations are given merely by the dimensions of the parallelepiped taken in different orders, all orientations (parallel to the L-R-V axes) are included on a single surface for each cavity. These surfaces are often discontinuous.

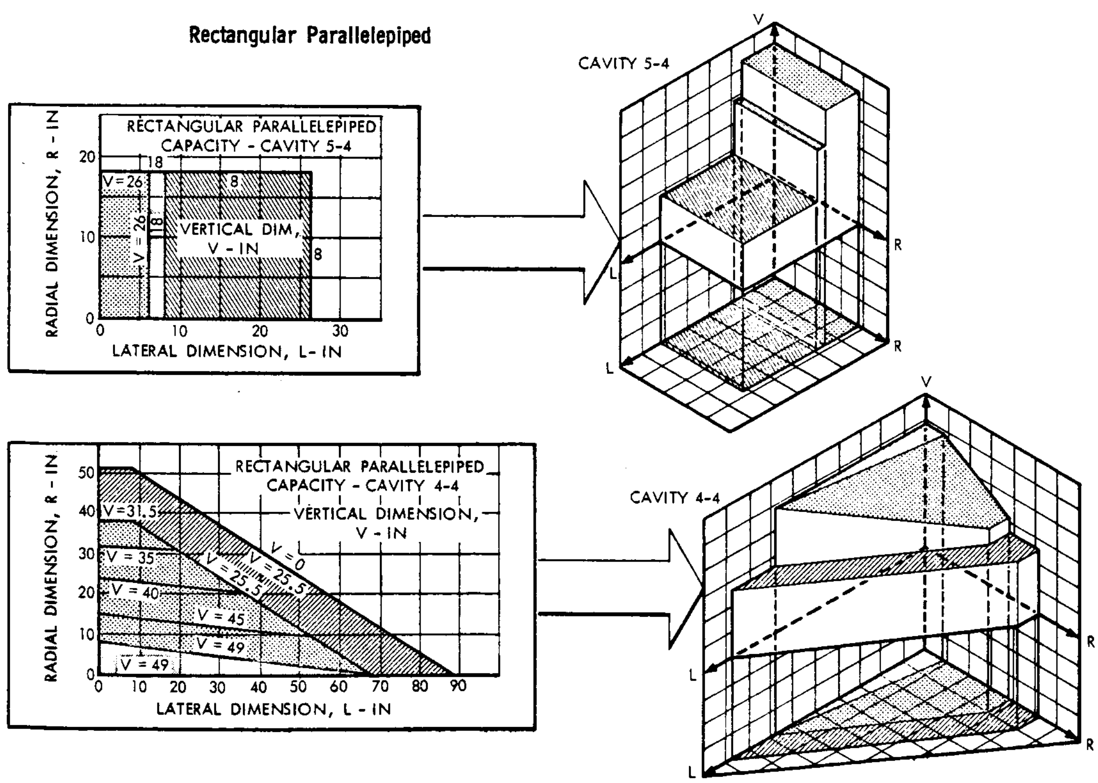


Figure 6-11 GEOMETRIC CAPACITIES COMPATIBILITY 2

It is again noted in Figure 6-11 that there is a distinct difference between the forms of the rectilinear and tapered capacities. The rectangular-parallelepiped capacity surfaces for rectilinear cavities are composed of planar horizontal rectangles (example cavity 5-4). These may be represented in a simpler manner. The surface in Figure 6-11 may be replaced by the dimensions for three rectangular parallelepipeds, at least one of which will contain any rectangular parallelepiped which can be contained in the cavity. The rectangular-parallelepiped capacity surfaces for tapered cavities (example cavity 4-4) are composed of inclined planes and curved surfaces of irregular shape. They cannot be simplified further.

It is interesting to note that the rectangular-parallelepiped capacity surface (shown in isometric form in Figure 6-11) is the original cavity distorted - beyond recognition in some cases - in such a way that its shape is simplified, its volume decreased, but its capacity for rectangular parallelepipeds maintained exactly.

The methodology used in the computer program to determine rectangular parallelepiped compatibility is illustrated in Figure 6-12.

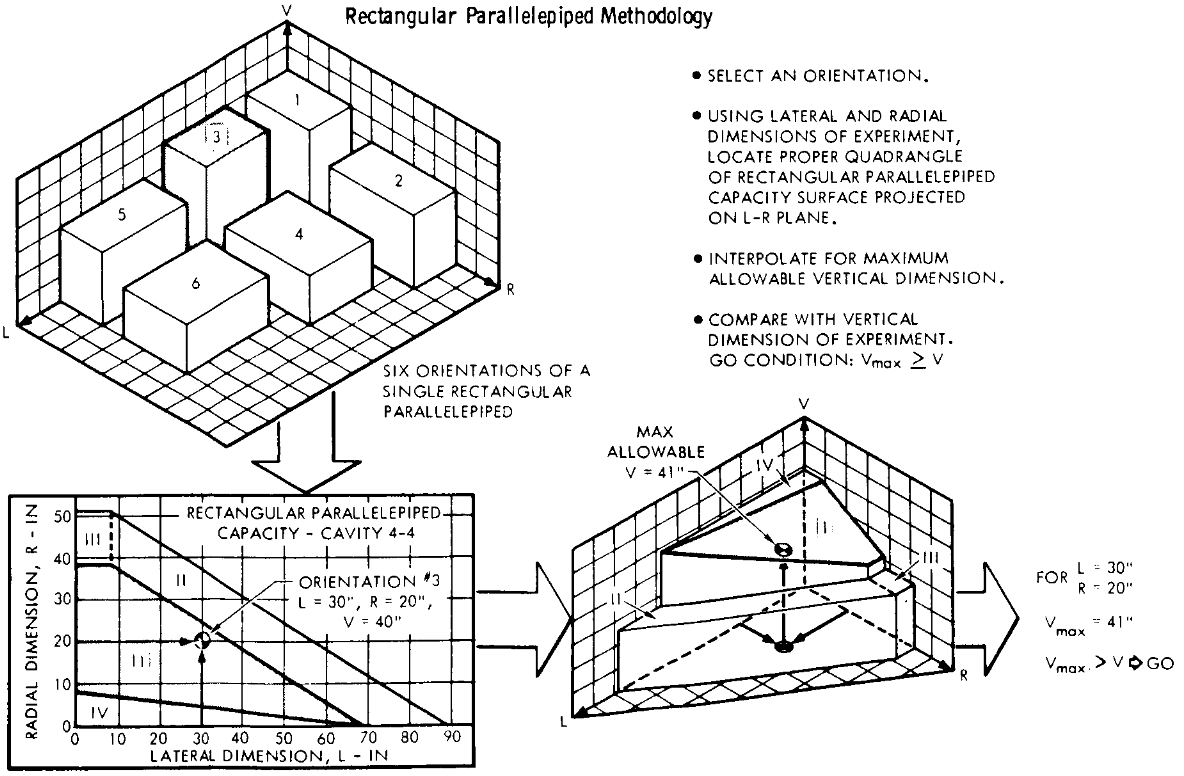


Figure 6-12 GEOMETRIC CAPACITIES COMPATIBILITY 3

The six possible orientations of a given parallelepiped within a cavity in which its sides are parallel to the axes of the cavity are shown. The example given is a rectangular parallelepiped measuring 20 by 30 by 40 inches. If the orientation of a rectangular parallelepiped payload is not critical, each of the six orientations is tried until one is found which will allow the payload to be contained within the cavity. In some cases, there may be a required alignment for only one axis of the payload, e.g., a camera pointing outboard. There are nine possible alignments of this type, each corresponding to a distinct pair of orientations. The orientation pairs given in the following table apply to the example given in Figure 6-12. In this case, only two orientations must be tried in order to determine compatibility. When the alignment of two axes is specified, there is only one possible orientation to be checked for compatibility.

Table of Orientation Pairs

Cavity Axes	Payload Axes: 20"	30"	40"
L	1, 2	3, 4	5, 6
R	3, 5	1, 6	2, 4
V	4, 6	2, 5	1, 3

Once an orientation is selected, the possibility of containing the rectangular parallelepiped with the selected orientation in a given cavity is determined in the following manner. The rectangular-parallelepiped capacity surface is divided into plane quadrangles. This does not mean that the cavity is being approximated as having planar faces, because the rectangular parallelpiped capacity surfaces are very nearly planar even for cavities possessing extreme curvature in their faces. These plane quadrangles are then projected onto the L-R plane. An equation involving the coordinates of the vertices of the quadrangles is applied in the computer program to the L and R dimensions of the payload in order to determine which of the quadrangles in the L-R plane contains the point represented by these dimensions. This operation indicates which plane must be interpolated on to determine the maximum possible vertical dimensions corresponding to those lateral and radial dimensions. When this dimension is found, a comparison is made with the actual vertical dimension of the experimental payload. It must be less than or equal to the maximum possible vertical dimension to be contained in the cavity. If the payload cannot be contained and another orientation is possible, the procedure is repeated with the next orientation.

6.2.6 Mode I Output

The output of Mode I is in the form of printed results and, if specified, an experimental payload-mission/vehicle compatibility library deck containing the required input for operation of Mode II. Although other problem options can be specified, the normal printed results consist of the following types and pages of data for each problem. Example results are given in the figures indicated:

1. Title page for Mode I and problem control input data (problem control options and optional overrides for the Mission/Vehicle/Primary Payload Characteristics Library). The example output given in Figure 6-13 identifies the mission and vehicle and library tape. In the example, all available problem control options were selected: (1) the determination of experiment/mission effectiveness, (2) the determination of experimental payload-mission/vehicle compatibility, (3) the generation of a Mode II compatibility library deck, and (4) the deletion of cavities originally included in the library. All available optional library overrides were also selected in the example problem: (1) launch date (a new solar declination was computed as a result of this override), (2) launch time (a new inclination to the terminator was computed as a result of this override), (3) vehicle-primary payload separation time, (4) primary mission duration, and (5) excess payload capability.

2. Experimental payload description input data obtained from the Experimental Payload Characteristics Library. The example output given in Figure 6-14 is a description of the operational requirements and physical characteristics of the experimental payload designated MS-3. The description data for this experimental payload are typical. However, the print-out format and data will vary with other experimental payloads, depending upon the applicability of each set of parameters to the specific payload. For example, a deployment time may be applicable in some cases. For experimental payloads which are ejected with an impulsive ΔV at in-plane and out-of-plane angles, these data are given in the output. Electromagnetic data may not be applicable for all experimental payloads; the corrective mass penalties computation for acoustics and vibration may be optionally specified; the heat dissipation rate or total heat dissipation thermal parameters may be optionally specified; the thermal parameters are not applicable in the orbit phase for ejected experimental payloads; the standard shape of the payload may

P R O G R A M   S E P T E R

S A T U R N   E X P E R I M E N T A L   P A Y L O A D

T E C H N I C A L   E V A L U A T I O N   A N D   R A T I N G

M O D E   I

S I N G L E   E X P E R I M E N T

C O M P A T I B I L I T Y   A N D   E F F E C T I V E N E S S

A N A L Y S I S

FLIGHT SA-207

FROM LIBRARY TAPE 001437A01

OVERRIDDEN MISSION/VEHICLE DATA

NEW LAUNCH DATE	31.0 MAR 1967
NEW SOLAR DECLINATION	4.2 DEG
NEW LAUNCH TIME	1000 EST
NEW INCLINATION TO TERMINATOR	152.8 DEG
NEW PRIMARY PAYLOAD SEPARATION TIME	5300. SEC
NEW PRIMARY MISSION DURATION	7.0 DAYS
NEW EXCESS PAYLOAD CAPABILITY	1000. LB

CAVITIES TO BE DELETED

1- 1   2- 4   6- 1

FOR EACH EXPERIMENTAL PAYLOAD--
MISSION/EFFECTIVENESS WILL BE DETERMINED
MISSION/VEHICLE COMPATIBILITY WILL BE DETERMINED

A SEPTER-MODE II LIBRARY DECK WILL BE GENERATED

Figure 6-13 SEPTER - MODE 1: TITLE AND PROBLEM CONTROL DATA

INFLIGHT EXPERIMENTAL PAYLOAD DESCRIPTION
EXPERIMENT MS- 3

AVAILABLE 1.0 JAN 1967 INSTALLATION TIME 5.0 DAYS

DEPLOYMENT MODE 1

ELECTROMAGNETIC DATA

	BAND	LOW FREQ (MC)	HIGH FREQ (MC)	LEVEL (DBM)
SENSITIVITY				
	1	1.00	230.00	-10.0
	2	230.00	240.00	-30.0
	3	240.00	250.00	-90.0
	4	250.00	260.00	-30.0
	5	260.00	10000.00	-10.0
SIGNALS				
	1	1.00	240.00	-40.0
	2	240.00	250.00	40.0
	3	250.00	750.00	-20.0
	4	750.00	10000.00	-40.0

ACOUSTICS DATA
NOISE TOLERANCE 150.0 DB SUSCEPTIBLE COMPONENTS MASS 63.3 LB
CORRECTIVE MASS PENALTIES MAY BE COMPUTED.

VIBRATIONS DATA
CORNER FREQUENCIES 10.0, 75.0 CPS TOLERANCE LEVELS 0.4, 10.0 G
SUSCEPTIBLE COMPONENTS MASS 63.3 LB
CORRECTIVE MASS PENALTIES MAY BE COMPUTED.

THERMAL DATA	PAD	LAUNCH	ORBIT
LOW TEMPERATURE TOLERANCE (DEG-F)	14.0	0.	-50.0
HIGH TEMPERATURE TOLERANCE (DEG-F)	75.0	240.0	65.0
HEAT DISSIPATION RATE (BTU/HR)	27.3	27.3	232.0
TOTAL HEAT DISSIPATION (BTU)			

PAYLOAD MASS 154.3 LB, VOLUME 3750. CU.IN., SHAPE REC.PAR.
TYPE AMORPHOUS

DIMENSIONS (IN)	LENGTH 3.0	WIDTH 10.0	HEIGHT 14.0
ALIGNMENT	NONE	NONE	NONE

DEVELOPMENT TIME 9.0 MO, COST $ 545300., RELIABILITY 0.9600

Figure 6-14 SEPTER - MODE 1: IN-FLIGHT EXPERIMENTAL PAYLOAD DESCRIPTION DATA be either a rectangular parallelepiped, a cylinder, or a sphere, and its form may be specified as either fixed or amorphous. In the case of rectangular parallelepiped or cylinder-shaped experimental payloads, the alignment of each dimension may be specified with respect to the vehicle axes (longitudinal, radial, or lateral). Development time, cost, and reliability data may not be available in some cases.

3. Experiment mission effectiveness array input data are given in the example output of Figure 6-15. These data are obtained from the Experimental Payload Characteristics Library. In the example, the maximum possible effectiveness is given as 100 percent. This is an input value determined by the effectiveness analyses method discussed in section 5. The effectiveness data are given in either one or two dimensional array tables. The example is a two-dimensional array table. The "key" row identifies (1) the array table number (up to 10 may be loaded, within a 25 row by 15 column restriction), (2) the origin of each table (coordinates of the lower left element of the table; in the example the origin is row five column one), (3) the size of the array (in the example there are seven values of x and four values of y), (4) the identification number "5" of the orbital elements which affect effectiveness (in the example the elements are numbers 10 and 11 of the order listed in Figure 6-16), and (5) the interpolation option selected (linear or fourth-order Lagrange). In the example the fourth-order Lagrange interpolation was selected. The effectiveness values in the array table of Figure 6-15 are given in rows one through four and columns two through eight for each x-y coordinate.

4. Experimental payload/mission effectiveness data. The example data given in Figure 6-16 were computed for an experimental payload (designated MS-3) which was not ejected from the vehicle (deployment Mode 0). Therefore, the mission parameters and orbital elements were computed for launch and ejection conditions of the primary payload. In the case of propulsive deployment of an experimental payload, the input values of ΔV and thrust angles are also given in the output. The double asterisks identify the mission parameters or orbital elements which were used to determine experiment effectiveness factors and the absolute experiment effectiveness. The value of normalized effectiveness is the ratio of absolute/maximum effectiveness.

MISSION EFFECTIVENESS ARRAY
EXPERIMENT MS- 3

MAXIMUM POSSIBLE EFFECTIVENESS 100.0 PERCENT

KEY	TABLE NO.	ORIGIN (X,Y)	SIZE (X)X(Y)	ELEMENTS (X)	(Y)	INTERP OPTION
	1	5, 1	7X 4	10	11	2

	1	2	3	4	5	6	7	8
1	2.00+00	7.10-01	8.64-01	8.80-01	8.90-01	8.93-01	8.97-01	9.00-01
2	1.60+00	5.20-01	8.30-01	8.90-01	9.10-01	9.20-01	9.27-01	9.30-01
3	1.20+00	2.40-01	5.30-01	8.40-01	9.50-01	9.68-01	9.80-01	9.90-01
4	1.00+00	3.00-02	3.10-01	5.80-01	9.30-01	9.75-01	9.90-01	1.00+00
5	0.	1.48+02	1.67+02	1.85+02	2.22+02	2.59+02	2.96+02	3.33+02
6	0.	0.	0.	0.	0.	0.	0.	0.
7	0.	0.	0.	0.	0.	0.	0.	0.
8	0.	0.	0.	0.	0.	0.	0.	0.
9	0.	0.	0.	0.	0.	0.	0.	0.
10	0.	0.	0.	0.	0.	0.	0.	0.
11	0.	0.	0.	0.	0.	0.	0.	0.
12	0.	0.	0.	0.	0.	0.	0.	0.
13	0.	0.	0.	0.	0.	0.	0.	0.
14	0.	0.	0.	0.	0.	0.	0.	0.
15	0.	0.	0.	0.	0.	0.	0.	0.
16	0.	0.	0.	0.	0.	0.	0.	0.
17	0.	0.	0.	0.	0.	0.	0.	0.
18	0.	0.	0.	0.	0.	0.	0.	0.
19	0.	0.	0.	0.	0.	0.	0.	0.
20	0.	0.	0.	0.	0.	0.	0.	0.
21	0.	0.	0.	0.	0.	0.	0.	0.
22	0.	0.	0.	0.	0.	0.	0.	0.
23	0.	0.	0.	0.	0.	0.	0.	0.
24	0.	0.	0.	0.	0.	0.	0.	0.
25	0.	0.	0.	0.	0.	0.	0.	0.

Figure 6-15 SEPTER - MODE 1: MISSION EFFECTIVENESS ARRAY DATA

EXPERIMENTAL PAYLOAD/MISSION EFFECTIVENESS
EXPERIMENT MS- 3

DEPLOYMENT PARAMETERS

MODE	TIME (SEC)	THETA (DEG)	PHI (DEG)	DELTA V (KM/SEC)
0	604.77	-0.	-0.	-0.

MISSION PARAMETERS AND ORBITAL ELEMENTS

	SEMIMAJOR AXIS	=	6564.05	KM
	ECCENTRICITY	=	0.0046	-
	INCLINATION	=	30.00	DEG
	ARGUMENT OF PERIGEE	=	80.78	DEG
	TRUE ANOMALY	=	2.01	DEG
	TIME OF PERIGEE PASS.	=	5263.36	SEC
	PERIGEE LATITUDE	=	29.57	DEG
	PERIOD	=	5292.57	SEC
	APOGEE ALTITUDE	=	215.95	KM
**	PERIGEE ALTITUDE	=	155.81	KM
**	APOGEE/PERIGEE ALT.	=	1.39	-
	LONG. OF NODAL PASS.	=	200.38	DEG
	TIME OF NODAL PASSAGE	=	4083.44	SEC
	INCLINATION TO TERM.	=	140.59	DEG
	SOLAR DECLINATION	=	23.37	DEG
	LAUNCH MONTH	=	6	-
	LAUNCH YEAR	=	1967	-
	JULIAN DATE	=	2439656.5	-
	MISSION DURATION	=	14.00	DAYS
	LAUNCH TIME	=	9.00	HR

EFFECTIVENESS PARAMETERS

EFFECTIVENESS FACTORS

0.525

EXPERIMENT EFFECTIVENESS (PCT)

ABSOLUTE E	MAXIMUM EMAX	NORMALIZED E/EMAX
52.5	100.0	52.5

Figure 6-16 SEPTER - MODE 1: EXPERIMENTAL PAYLOAD/MISSION EFFECTIVENESS, MISSION PARAMETERS, AND ORBITAL ELEMENTS

5. Experimental payload/vehicle compatibility. The example data given in Figure 6-17 indicate the GO/NO-GO compatibility of the experimental payload MS-3 with complete-vehicle or vehicle-zone dependent parameters. Warning-type statements are given in the case of an availability date/launch date incompatibility and electromagnetic interference (EMI). Incompatibilities for these parameters do not affect the final NO-GO decision for an experimental payload. The acoustics and vibration data reflect the computed mass penalties required in order to correct tolerance deficiencies. These penalties affect the overall NO-GO and final NO-GO decisions only if their masses are equal to or greater than the mass of the susceptible components of the experimental payload.

6. Experimental payload/cavity compatibility data. These data indicate the GO/NO-GO compatibility for each cavity dependent criterion. The overall decision for each criteria group, e.g., environmental compatibility, is identified by an asterisk. The overall decisions affect the final GO/NO-GO decision. In the example given in Figure 6-18, the deployment compatibility is only dependent on the deployment mode. A deployment time compatibility is indicated in the case of ejected experimental payloads. In the case of the thermal parameters compatibility, only the applicable mission phases are given in the print-out. If an incompatibility is calculated, each thermal parameter is listed with its corresponding GO/NO-GO decision. The geometric compatibility data are given for all specified alignments. In the example, no alignment was specified. Therefore, all six possible alignments were tried for the example rectangular parallelepiped. As noted, the final decision is GO for the example experimental payload and specified cavity. Experimental payload/cavity compatibility data are given for each payload-cavity combination. The results calculated for another example experimental payload and cavity are shown in Figure 6-19. In this case the final decision is NO-GO because the experimental payload mass exceeds the structural mass limit of the cavity.

7. Experimental payload/mission/vehicle compatibility summary. The results given in the example output of Figure 6-20 summarize the GO/NO-GO compatibility of one experimental payload utilizing all available cavities. The compatibility is given for each criterion. Those criteria that do not affect the overall decision are listed in the top right of the print-out. The value of the normalized experiment/mission effectiveness is also given on the summary page of output.

EXPERIMENTAL PAYLOAD/VEHICLE COMPATIBILITY

EXPERIMENT MS- 3
VEHICLE SA-207

AVAILABILITY DATE/LAUNCH DATE BUFFER = 160. DAYS GO

ELECTROMAGNETIC INTERFERENCE

POSSIBLE EXPERIMENT-VEHICLE INTERFERENCE ON FOLLOWING BANDWIDTHS

9.90 TO 10.10 MC
230.00 TO 305.00 MC
2050.00 TO 2300.00 MC

POSSIBLE VEHICLE-EXPERIMENT INTERFERENCE ON FOLLOWING BANDWIDTHS

10.00 TO 10.50 MC
225.00 TO 295.00 MC
2200.00 TO 2300.00 MC

ACOUSTICS DATA

AC/VIB ZONE	1	MASS PENALTY REQUIRED =	0.9	GO
AC/VIB ZONE	2	MASS PENALTY REQUIRED =	0.9	GO

VIBRATIONS DATA

AC/VIB ZONE	1	MASS PENALTY REQUIRED =	5.5	GO
AC/VIB ZONE	2	MASS PENALTY REQUIRED =	9.2	GO

Figure 6-17 SEPTER - MODE 1: EXPERIMENTAL PAYLOAD/VEHICLE COMPATIBILITY

EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

EXPERIMENT MS- 3
CAVITY 1- 1

DEPLOYMENT COMPATIBILITY

MODE	GO	*

ENVIRONMENTAL COMPATIBILITY

THERMAL		
PAD	GO	
LAUNCH	GO	
ORBIT	GO	
ACOUSTIC CORRECTIVE MASS PENALTY OF 0.9 ADDED	GO	
VIBRATION CORRECTIVE MASS PENALTY OF 5.5 ADDED	GO	
OVERALL DECISION	GO	*

MASS COMPATIBILITY

STRUCTURAL LIMIT = 1000.0 TOTAL EXP.MASS= 160.8	GO	*

GEOMETRIC COMPATIBILITY

AVAIL.VOL.=163966. REQ.VOL.= 3750. PCT.USED= 2.3	GO	
STANDARD SHAPE--RECTANGULAR PARALLELEPIPED		
L= 3.0 R= 10.0 V= 14.0	GO	
L= 3.0 R= 14.0 V= 10.0	GO	
L= 10.0 R= 3.0 V= 14.0	GO	
L= 10.0 R= 14.0 V= 3.0	GO	
L= 14.0 R= 3.0 V= 10.0	GO	
L= 14.0 R= 10.0 V= 3.0	GO	
OVERALL DECISION	GO	*

****
FINAL * GO * DECISION
****

Figure 6-18 SEPTER - MODE 1: EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

```
        EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

                    EXPERIMENT  MI- 1
                        CAVITY   3- 6

DEPLOYMENT COMPATIBILITY

   MODE                                                          GO  *
   TIME                                                          GO  *


ENVIRONMENTAL COMPATIBILITY

   THERMAL
      PAD                                                        GO
      LAUNCH                                                     GO

   ACOUSTIC   CORRECTIVE MASS PENALTY OF    5.8 ADDED            GO

   VIBRATION  CORRECTIVE MASS PENALTY OF   35.0 ADDED            GO

                                   OVERALL DECISION              GO  *


MASS COMPATIBILITY

   STRUCTURAL LIMIT = 1000.0  TOTAL EXP.MASS= 1122.5           NOGO *


GEOMETRIC COMPATIBILITY

    AVAIL.VOL.=171039.  REQ.VOL.= 81200.  PCT.USED=  47.5  GO

    STANDARD SHAPE--RECTANGULAR PARALLELEPIPED
               L= 44.0   R= 22.0   V= 31.0                      GO
               L= 44.0   R= 31.0   V= 22.0                      NOGO
               L= 22.0   R= 44.0   V= 31.0                      NOGO
               L= 22.0   R= 31.0   V= 44.0                      NOGO
               L= 31.0   R= 44.0   V= 22.0                      NOGO
               L= 31.0   R= 22.0   V= 44.0                      GO

                                   OVERALL DECISION              GO  *


                       ********
                 FINAL * NOGO * DECISION
                       ********
```

Figure 6-19 SEPTER - MODE 1: EXPERIMENTAL PAYLOAD/CAVITY COMPATIBILITY

EXPERIMENTAL PAYLOAD/MISSION/VEHICLE COMPATIBILITY SUMMARY
EXPERIMENT MS- 3/FLIGHT SA-207

NORMALIZED MISSION EFFECTIVENESS 52.5 PERCENT AVAILABILITY GO
POSSIBLE EMI YES

EXPERIMENT/CAVITY COMPATIBILITY

CAVITY	DEPL MODE	DEPL TIME	THERMAL	ACOUS	VIB	MASS ATTACH	VOLUME	GEOM	OVERALL
1- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
1- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
1- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 4	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 5	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 6	GO	N/A	GO	GO	GO	GO	GO	GO	GO
2- 7	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 4	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 5	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 6	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 7	GO	N/A	GO	GO	GO	GO	GO	GO	GO
3- 8	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 1	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 2	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 3	GO	N/A	GO	GO	GO	GO	GO	GO	GO
4- 4	GO	N/A	GO	GO	GO	GO	GO	GO	GO
5- 1	GO	N/A	NOGO	GO	GO	NOGO	GO	GO	NOGO
5- 2	GO	N/A	NOGO	GO	GO	GO	NOGO	NOGO	NOGO
5- 3	GO	N/A	NOGO	GO	GO	GO	NOGO	NOGO	NOGO
5- 4	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 5	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 6	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 7	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
5- 8	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
6- 1	GO	N/A	NOGO	GO	GO	NOGO	NOGO	NOGO	NOGO
6- 2	GO	N/A	NOGO	GO	GO	NOGO	GO	GO	NOGO
6- 3	GO	N/A	NOGO	GO	GO	NOGO	NOGO	NOGO	NOGO
6- 4	GO	N/A	NOGO	GO	GO	NOGO	GO	GO	NOGO
6- 5	GO	N/A	NOGO	GO	GO	NOGO	NOGO	NOGO	NOGO
7- 1	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 2	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 3	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 4	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO
7- 5	GO	N/A	NOGO	GO	GO	GO	GO	GO	NOGO

Figure 6-20 SEPTER - MODE 1: EXPERIMENTAL PAYLOAD/MISSION/VEHICLE COMPATIBILITY SUMMARY

6.3 MODE II OPERATION

Mode II analysis consists of the arrangement of multiple experimental payloads in the vehicle so that (1) a preferred order (priority) loading is used in the arrangement according to externally prepared preference list(s), (2) no payload-vehicle or payload cavity incompatibilities exist, (3) the payload mass capability of the mission/vehicle is not exceeded, and (4) the near-maximum number of experimental payloads within the placement policy mechanics of the program are placed aboard the vehicle from the preference list. Mode II output, therefore, consists of an arrangement of experimental payloads aboard the vehicle in the preferred order with no incompatibilities.

6.3.1 Libraries

In the Mode II operation, definition-type library data are supplied as a part of the Mode I output. These data are largely the same as those provided by the Mission/Vehicle/Primary Payload Characteristics Library and the Experimental Payload Characteristics Library for Mode I, (Subsection 6.2.1). In Mode II, however, some data are deleted (e.g., mission characteristics) and other data are added as a result of computations completed in the Mode I operation. The additional library data which are of most significance for the Mode II operation are those which specify the compatible cavities for each experimental payload. Other additional data are the mass penalties calculated as a result of acoustics/vibration deficiencies for each experimental payload.

6.3.2 External Analysis

The external analysis required for the Mode II operation consists of compiling preference list(s). These lists are simply a tabulation of experimental payload identifications in a preferred order of loading in a given vehicle for a given mission. Although the compatibility and effectiveness output data of Mode I are obviously provided to assist the user in the formulation of preference lists, any additional data or methods of establishing priority may be used in arriving at a preference list. Several sets of preference lists may be formulated for a given set of experiments.

6.3.3 Problem Input and Controls

Inputs for the operation of Mode II consist of library data, problem data, and optional library overrides. The library data (card deck) is generated as output from the Mode I operation and contains (1) the vehicle, cavity, and experimental payload description data obtained from the Mission/Vehicle/Primary Payload Characteristics Library of Mode I and (2) computed compatibility data from Mode I, i.e., the identification of all cavities with which an experimental payload is compatible (final GO decision), mass penalties for acoustics and vibration tolerance deficiencies (if any), and any library override data which may have been used in Mode I. Problem data (card decks) consist of (1) the preference list which is prepared by the user of the program in order to establish the desired order (priority) in which experimental payloads are to be loaded aboard the vehicle, (2) options for print-out and placement policy (based on mass or volume), and (3) controls for placement policy iteration and cutoff. Optional library overrides may also be specified, e.g., excess payload capability and Mode I compatibility (including predetermined placements for arbitrary experiments).

6.3.4 Multiple Payload Arrangement Logic

The purpose of the multiple payload arrangement analysis is to determine the arrangements of payloads in cavities throughout the vehicle in such a manner that no incompatibilities occur within any cavity and that the payloads are loaded in a preferred order. Arrangements which will allow the greatest number of payloads within the overall mass and volume limits of the vehicle are the desired result.

The arrangement analysis is an optimization problem, but optimization methods (except for complete enumeration, which is not feasible because of the extremely large number of possible arrangements) are not readily applicable. Consideration of the problem indicates that optimal arrangements will not usually be unique. This conclusion is evident because the mass attachment limit for a cavity divided by its available volume is generally a smaller number than the densities of typical experiments. In addition, the sum of the cavity mass attachment limits is usually greater than the payload capability of the vehicle. This indicates that in loading the vehicle, mass limits will be encountered prior to volume limits. It is further implied that the maximum number of experiments which can be loaded will depend more on the payload capability of the vehicle than on the available volume or arrangement of the payloads in the vehicle. Consequently, an attempt to apply a true optimization process to the problem appears to involve a degree of effort not justified by the results desired from this study.

In Figure 6-21, a basic outline of an alternate approach aimed at satisfying all constraints and directly searching for one of the non-unique "optimal" solutions or arrangements is shown. The method consists of assuming that an arrangement can be found for the maximum possible number (on the basis of overall mission/vehicle constraints) of experimental payloads on the preference list. If an arrangement cannot be found for this maximum number, an experimental payload is dropped from the preference list, and the arrangement process is repeated. The approach is simple in concept, but its application is complex. The overall concept of the multiple payload arrangement logic, including the basic input requirements, the general placement operations, and the basic output is shown in Figure 6-21. The input requirements listed in Figure 6-21 are a summary of the problem input and controls discussed in subsection 6.3.3. The Mode II output is discussed in detail in subsection 6.3.5.

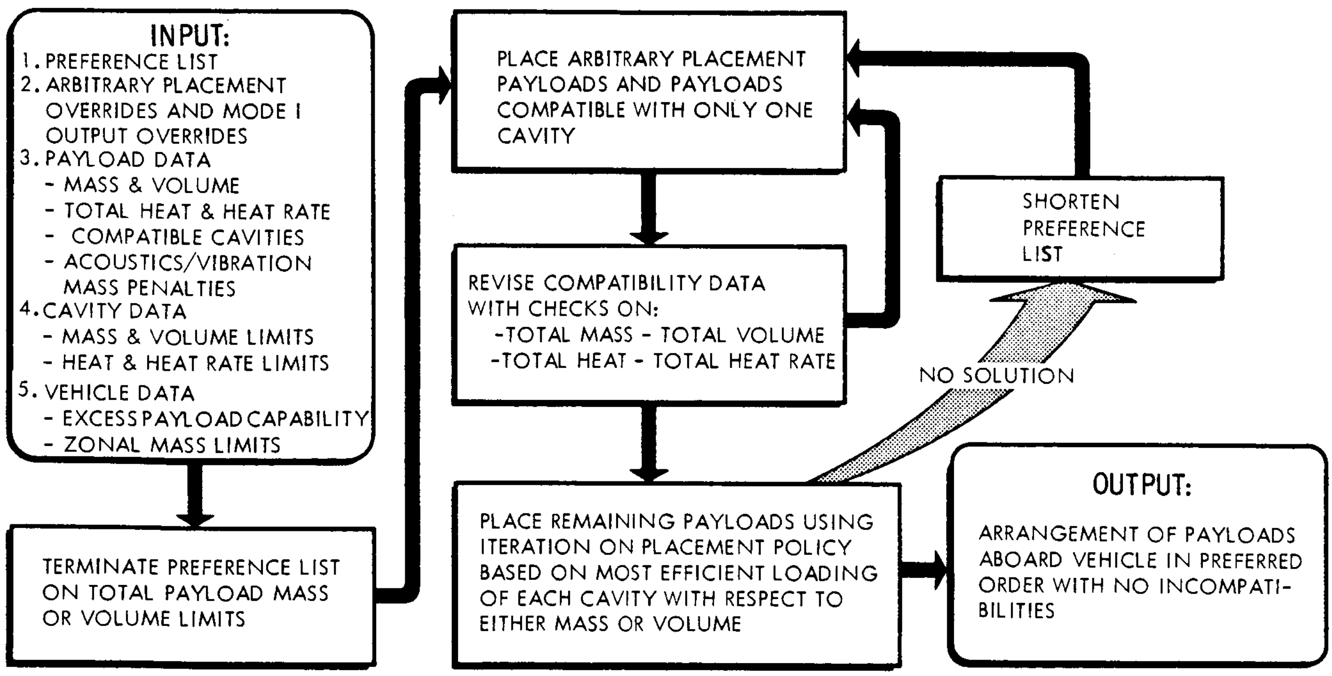


Figure 6-21 MULTIPLE PAYLOAD ARRANGEMENT LOGIC

The initial operations in the computer program logic are those performed to eliminate any incompatibilities in the experimental payload preference list and to terminate the preference list on the vehicle/mission total mass and volume limits. The arrangement logic consists of multiple iterations. However, the general placement operations may be separated into two basic iteration procedures: (1) the placement of experimental payloads which have no choice of location (i.e., arbitrary, as determined by an override option or compatibility, determined by Mode I analyses), (2) the placement of payloads which have a choice of location, based upon an arbitrary set of rules for placement (placement policy) of experimental payloads. If no solution is found in the second basic iteration, the preference list is shortened and the entire placement procedure is repeated.

6.3.4.1 Multiple Payload Arrangement Placement Policy Logic

The flow of calculations and iterations of the multiple payload arrangement placement policy logic is shown in Figure 6-22. This flow diagram is a more detailed illustration of the placement operations of the overall concept presented in Figure 6-21.

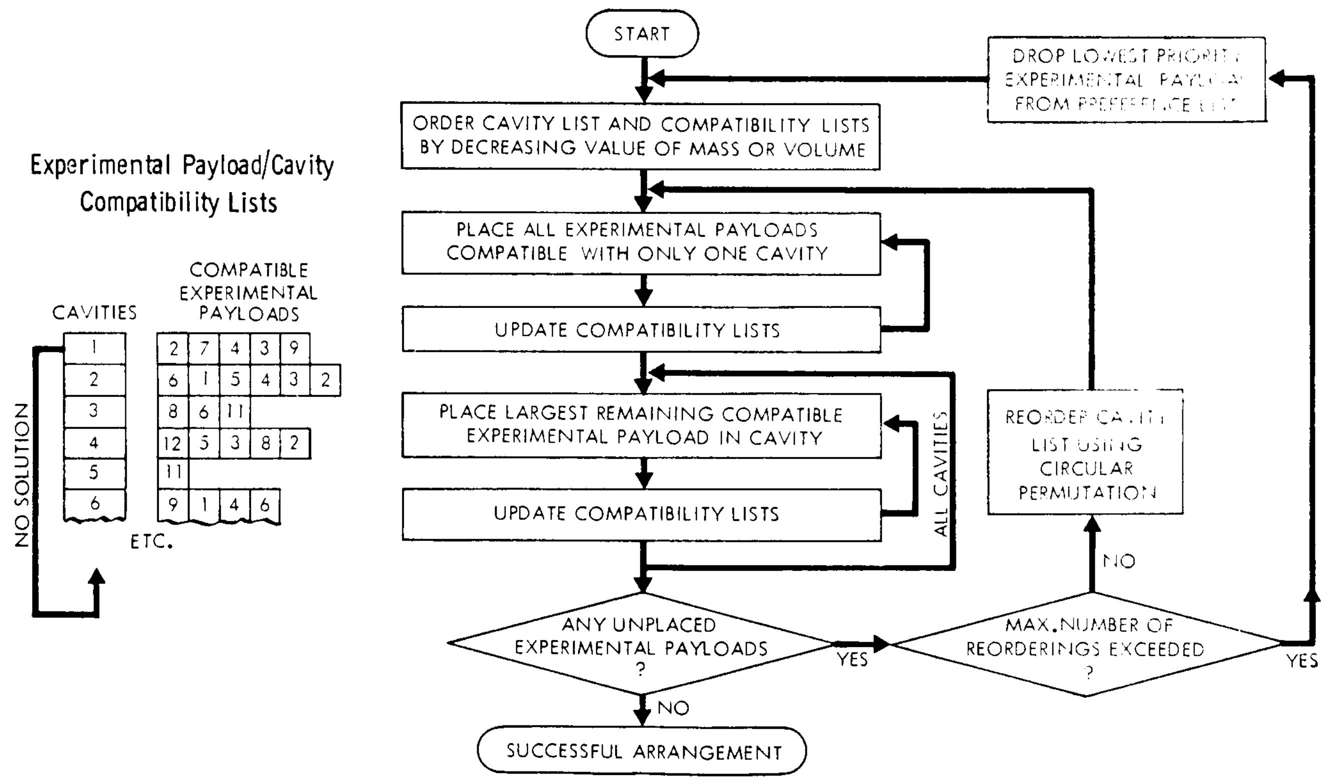


Figure 6-22 MULTIPLE PAYLOAD ARRANGEMENT PLACEMENT POLICY LOGIC

The initial operation performed in the placement policy logic is to order (list) the available cavities by decreasing mass or volume (depending upon the placement policy option selected). Compatible experimental payloads for each cavity are similarly ordered. An example listing of cavities and compatible experimental payloads is given in Figure 6-22.

After the ordering operation has been completed, experimental payloads compatible with only one cavity are placed. In the example, payload number seven is only compatible with cavity number one. The placement of this payload may, however, revise the compatibility list for cavity number one, i.e., some or all of the remaining payloads (numbers 2, 4, 3 and 9) formerly compatible with cavity number one may now be incompatible. Therefore, after every placement of a payload, the compatibility lists are updated in the computer program, and the iteration procedure which places payloads compatible with only one cavity is iterated until all remaining payloads have a choice of location.

The placement policy is applied to all experimental payloads having a choice of location. The placement policy logic in the computer program attempts to place in the largest remaining cavity the largest experimental payload which is compatible with the cavity. The term "largest" refers to either mass or volume (as applicable to the placement policy option selected). The term "remaining" refers to compatible experimental payloads which have not been placed or cavities which have not been filled, at any point in the placement computation. The placement policy is applied, as shown in Figure 6-22, cavity-by-cavity in their listed order. If all experimental payloads on the preference list are placed, the arrangement is successful and the procedure is terminated. If any unplaced experimental payloads remain, the cavity list is reordered by a circular permutation, i.e., cavity number one is placed at the bottom of the list. Iteration is required in this search of possible arrangements because the placement policy is arbitrary and is influenced by the order in which the cavities are listed. The placement procedure is repeated, using the new order of cavities, until either a successful arrangement is found or until the arbitrary specified maximum number of reorderings has been exceeded.

At the point where the maximum number of reorderings has been exceeded, it is assumed that no solution will be found for the number of experimental payloads remaining on the preference list. An arbitrary number of reorderings is specified because in order to ensure that the above assumption (no solution will be found) is true would require trying the placement procedure on every possible permutation of the cavity list combined with all possible sets of permutations of the compatible experimental payload lists. Computation times for even short lists are measured in centuries. Thus, only the cavity list is reordered, and an arbitrary number of circular permutations is used because it is convenient.

In the case where the maximum number of reorderings has been exceeded, the lowest priority experimental payload is dropped from the preference list, and the entire placement procedure is terminated either when a successful arrangement (all remaining experimental payloads on the preference list are placed) is found or when an arbitrary specified maximum number of arrangement attempts has been made. The maximum number of arrangement attempts are specified in order to allow the user of the program to terminate a problem within a reasonable desired computer running time.

6.3.5 Mode II Output

The output of Mode II is in the form of printed results consisting of the following types and pages of data for each problem (unless problem options specify otherwise). Example results are given in the figures indicated:

1. Title page for Mode II, mission identification, and library override input data (if any). The example output given in Figure 6-23 identifies the mission and vehicle, the launch date, and the excess payload capability. The excess payload is that value specified in the library or an override value.

2. Problem control and preference list data. Example data are given in Figure 6-24. All problem options and controls must be specified: (1) the placement policy (based on mass or volume), (2) the placement policy iteration (number of times the cavity list is reordered before an experimental payload is dropped from the preference list, and (3) the placement policy cutoff (maximum number of arrangements to be attempted in the event that no arrangement solution is found prior to the specified cutoff). The example preference list is the desired order of loading (priority) of 30 experimental payloads aboard the vehicle.

3. Identification of experimental payloads dropped from the preference list. An example of this type of output is given in Figure 6-25. In the example, experimental payloads were dropped from the preference list because they are incompatible with all cavities used in the problem. However, experimental payloads may be dropped for any of the following reasons: (1) an experimental payload is not identified in the library, (2) an experimental payload is incompatible with all cavities used, (3) the mass of a single experimental payload exceeds the excess payload capability of the mission/vehicle, (4) the total mass or volume of all higher priority experimental

P R O G R A M   S E P T E R

S A T U R N   E X P E R I M E N T A L   P A Y L O A D

T E C H N I C A L   E V A L U A T I O N   A N D   R A T I N G

M O D E   I I

M U L T I P L E   E X P E R I M E N T

C O M P A T I B I L I T Y   A N D   A R R A N G E M E N T

A N A L Y S I S

FLIGHT SA-207

LAUNCH 15.0 JUN 1967

EXCESS PAYLOAD CAPABILITY   10000.0 LB

Figure 6-23 SEPTER - MODE II: TITLE AND MISSION IDENTIFICATION DATA

PREFERENCE LIST NUMBER 1

THE PLACEMENT POLICY WILL BE BASED ON MASS. THE CAVITIES WILL BE REORDERED 22 TIMES BEFORE THE PREFERENCE LIST IS SHORTENED. A MAXIMUM OF 88 ARRANGEMENTS WILL BE ATTEMPTED. THE FOLLOWING IS THE PREFERRED ORDER OF PLACEMENT.

PREFERENCE	EXPERIMENT
1	SDT- 1
2	SDT- 2
3	SDT- 3
4	SDT- 4
5	SDT- 5
6	MS- 1
7	MS- 2
8	MS- 3
9	MS- 4
10	MS- 5
11	MI- 1
12	MI- 2
13	MI- 3
14	MI- 4
15	MI- 5
16	M- 1
17	M- 2
18	M- 3
19	M- 4
20	M- 5
21	OEA- 1
22	OEA- 2
23	OEA- 3
24	OEA- 4
25	OEA- 5
26	SLG- 1
27	SLG- 2
28	SLG- 3
29	SLG- 4
30	SLG- 5

Figure 6-24 SEPTER - MODE II: PROBLEM CONTROL AND PREFERENCE LIST DATA

```
EXPERIMENT  MS- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT  MI- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT  MI- 3 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT  MI- 5 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT OFA- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT SLG- 1 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT SLG- 3 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.

EXPERIMENT SLG- 4 HAS BEEN DROPPED FROM THE PREFERENCE LIST.
   THIS EXPERIMENT IS NOT COMPATIBLE WITH ANY CAVITY.
```

Figure 6-25 SEPTER - MODE II: IDENTIFICATION OF INCOMPATIBLE EXPERIMENTAL PAYLOADS

payloads exceeds the excess payload capability, or the total volume of all cavities used, or the total mass attachment limit for all cavities used, and (5) no solution for an arrangement has been found for the specified number of cavity reorderings.

4. Compatibility array data. Example data are given in Figure 6-26. The experimental payload preference list and the cavities with which each payload is compatible or incompatible (as determined from the Mode I operation) are shown in a compatibility array. The number and designation of cavities with which all of the listed experimental payloads are incompatible are also given in the output.

5. Description of multiple experimental payload arrangements (by cavities). Example arrangement description data are given in Figure 6-27. If the print-out indicates that there are no unplaced experimental payloads, the solution for an arrangement of all experimental payloads on the preference list is successful. The data given in Figure 6-27 are the identity of the experimental payloads contained in each cavity; the total values of mass, volume, and thermal parameters contained in each cavity; and the remaining values. The structural group to which each cavity is assigned is also identified.

6. Summary table of experimental payload arrangements. The summary table may indicate successful or unsuccessful arrangements. An example of a successful arrangement (i.e., there are no unplaced experimental payloads remaining) is given in Figure 6-28. The experimental payloads are listed according to their rank in the preference list, and the cavity in which each experimental payload is contained (if placed) is identified. Experimental payloads that have not been placed and vacant cavities are also identified.

6.4 REFERENCES

6.1 <u>Preliminary Definition of Saturn Instrument Unit and S-IVB Support Capabilities for Extended Apollo Earth-Orbit Experiments</u>, NASA/MSFC Publication, 15 April 1965 (U)

COMPATIBILITY ARRAY

THIS COMPATIBILITY ARRAY WILL BE USED IN DETERMINING PLACEMENTS

(XX COMPATIBLE 00 INCOMPATIBLE)

CAVITY ZONE AND NUMBER

PREF	EXP	1	1	1	2	2	2	2	2	2	2	3	3	3	3	3	3	3	3	4	4	4	4
		1	2	3	1	2	3	4	5	6	7	1	2	3	4	5	6	7	8	1	2	3	4
1	SDT- 1	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	00
2	SDT- 2	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
3	SDT- 3	XX	XX	XX	00	XX	00	00	00	00	00	00	XX	00	XX	00	XX	00	XX	00	00	00	00
4	SDT- 4	XX	XX	XX	00	XX	00	00	00	00	00	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
5	SDT- 5	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	00
6	MS- 2	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
7	MS- 3	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
8	MS- 4	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
9	MS- 5	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
10	MI- 2	XX	XX	XX	XX	XX	XX	XX	00	00	XX	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
11	MI- 4	XX	XX	XX	00	XX	00	00	00	00	00	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
12	M- 1	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
13	M- 2	XX	XX	XX	00	00	00	00	00	00	00	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
14	M- 3	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX
15	M- 4	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	XX
16	M- 5	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX
17	OEA- 2	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	00	XX	XX	XX	XX
18	OEA- 3	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	00	00	00	00
19	OEA- 4	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	00	XX	00	XX	00	XX	00	00	00	00
20	OEA- 5	XX	XX	XX	XX	XX	XX	XX	00	00	XX	00	XX	00	XX	00	XX	00	XX	XX	XX	XX	XX
21	SLG- 2	XX	XX	XX	00	00	00	00	00	00	00	XX	XX	XX	XX	XX	XX	XX	XX	00	00	00	00
22	SLG- 5	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	00	XX	XX	XX	XX	XX	XX	XX

28 INCOMPATIBLE CAVITIES

5- 1	5- 2	5- 3	5- 4	5- 5	5- 6	5- 7	5- 8
6- 1	6- 2	6- 3	6- 4	6- 5	7- 1	7- 2	7- 3
7- 4	7- 5	7- 6	7- 7	7- 8	7- 9	7-10	7-11
7-12	7-13	7-14	7-15				

Figure 6-26 SEPTER - MODE II: COMPATIBILITY ARRAY-INCOMPATIBLE CAVITIES

```
              MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT DESCRIPTION

                 0 UNPLACED EXPERIMENTS        NUMBER    5

CAVITY  1- 1       STRUCTURAL GROUP  1
  CONTAINS  2 EXPS. SDT- 2  SDT- 3

  TOTAL MASS =     891.9 LB       TOTAL VOLUME =    76589. CU.IN.
  REMAINDER  =     108.1 LB       REMAINDER     =   87377. CU.IN.

                                  PAD   LAUNCH   ORBIT
  HEAT DISSIPATION RATE            0.       0.      0.  BTU/HR
  REMAINDER                      200.0   100.0   300.0 BTU/HR
  TOTAL HEAT DISSIPATION           0.       0.      0.  BTU
  REMAINDER                       17.0    17.0    17.0 BTU

CAVITY  1- 2       STRUCTURAL GROUP  2
  CONTAINS  3 EXPS. SDT- 4   MS- 2    M- 1

  TOTAL MASS =     889.2 LB       TOTAL VOLUME =    67983. CU.IN.
  REMAINDER  =     110.8 LB       REMAINDER     =   95983. CU.IN.

                                  PAD   LAUNCH   ORBIT
  HEAT DISSIPATION RATE           27.3    27.3   225.0 BTU/HR
  REMAINDER                      172.7    72.7    75.0 BTU/HR
  TOTAL HEAT DISSIPATION           0.       0.      0.  BTU
  REMAINDER                       17.0    17.0    17.0 BTU

CAVITY  1- 3       STRUCTURAL GROUP  3
  CONTAINS  4 EXPS.  MS- 3    M- 2    M- 3    M- 5

  TOTAL MASS =     806.7 LB.      TOTAL VOLUME =    28123. CU.IN.
  REMAINDER  =     193.3 LB       REMAINDER     =  135843. CU.IN.

                                  PAD   LAUNCH   ORBIT
  HEAT DISSIPATION RATE           67.9    67.9   249.1 BTU/HR
  REMAINDER                      132.1    32.1    50.9 BTU/HR
  TOTAL HEAT DISSIPATION           0.       0.      0.  BTU
  REMAINDER                       17.0    17.0    17.0 BTU
```

Figure 6-27 SEPTER - MODE II: MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT DESCRIPTION

MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT SUMMARY 5

(XX CONTAINED IN -- NOT CONTAINED IN 00 UNPLACED)

		CAVITY ZONE AND NUMBER												
		1	1	1	2	2	2	3	3	3	3	3	4	4
		1	2	3	1	5	7	1	2	4	6	8	1	2
PREF	EXP													
1	SDT- 1	--	--	--	--	--	--	--	--	--	--	--	XX	--
2	SDT- 2	XX	--	--	--	--	--	--	--	--	--	--	--	--
3	SDT- 3	XX	--	--	--	--	--	--	--	--	--	--	--	--
4	SDT- 4	--	XX	--	--	--	--	--	--	--	--	--	--	--
5	SDT- 5	--	--	--	--	--	--	--	--	--	--	--	--	XX
6	MS- 2	--	XX	--	--	--	--	--	--	--	--	--	--	--
7	MS- 3	--	--	XX	--	--	--	--	--	--	--	--	--	--
8	MS- 4	--	--	--	XX	--	--	--	--	--	--	--	--	--
9	MS- 5	--	--	--	--	XX	--	--	--	--	--	--	--	--
10	MI- 2	--	--	--	--	--	--	--	XX	--	--	--	--	--
11	MI- 4	--	--	--	--	--	--	--	--	XX	--	--	--	--
12	M- 1	--	XX	--	--	--	--	--	--	--	--	--	--	--
13	M- 2	--	--	XX	--	--	--	--	--	--	--	--	--	--
14	M- 3	--	--	XX	--	--	--	--	--	--	--	--	--	--
15	M- 4	--	--	--	--	--	--	--	--	--	--	--	XX	--
16	M- 5	--	--	XX	--	--	--	--	--	--	--	--	--	--
17	OEA- 2	--	--	--	--	--	--	--	--	--	--	--	--	XX
18	OEA- 3	--	--	--	XX	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
19	OEA- 4	--	--	--	--	--	--	--	--	--	XX	--	--	--
20	OEA- 5	--	--	--	--	--	--	--	--	--	--	XX	--	--
21	SLG- 2	--	--	--	--	--	--	XX	--	--	--	--	--	--
22	SLG- 5	--	--	--	--	--	XX	--	--	--	--	--	--	--

37 VACANT CAVITIES

2- 2	2- 3	2- 4	2- 6	3- 3	3- 5	3- 7	4- 3
4- 4	5- 1	5- 2	5- 3	5- 4	5- 5	5- 6	5- 7
5- 8	6- 1	6- 2	6- 3	6- 4	6- 5	7- 1	7- 2
7- 3	7- 4	7- 5	7- 6	7- 7	7- 8	7- 9	7-10
7-11	7-12	7-13	7-14	7-15			

Figure 6-28 SEPTER - MODE II: MULTIPLE EXPERIMENTAL PAYLOAD ARRANGEMENT SUMMARY

6.2 Saturn IB Payload Planner's Guide, Douglas Report SM-47010, Missile and Space Systems Division, Huntington Beach, California, March 1965 (U)

6.3 The Study of the Utilization of the Saturn IB Instrument Unit to Support Space Experiments, ASTRAN Report 8-11301-AR, 3 November 1964 (U)

part III APPENDICES

APPENDIX A

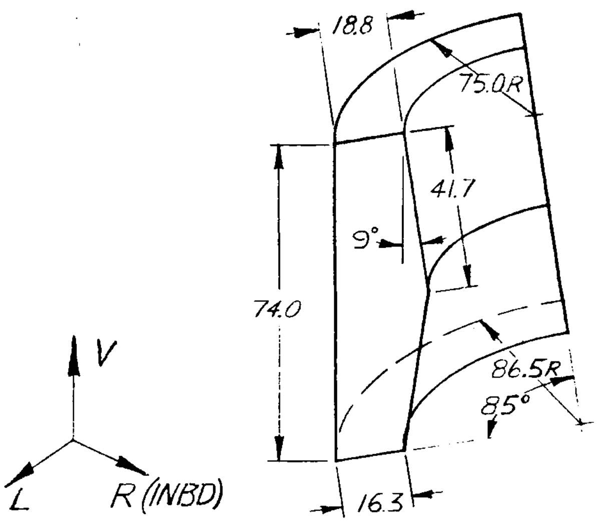
CAVITY DESCRIPTIONS

This appendix contains descriptive drawings and standard shapes capacity curves for the 53 cavities defined in the Saturn In-Flight Experimental Payload Study. The location of each of these cavities is shown in Figure 3-3 in the body of the report. The dimensions and orientation of the cavity are shown on the descriptive drawing. In addition, the cavity location, effectivity, mass capacity, and sphere capacity are also defined. The V, R, and L axes system is used in defining the orientations of the cavities. The V axis is generally parallel to the launch vehicle longitudinal axis, the R axis is generally normal to the external contour of the vehicle, and the L axis is 90 degrees to both the V and R axes.

The capacity of the cavity to contain the standard geometrical shapes, parallelepiped and cylinder, is shown in the standard shape capacity curves. The capacity to contain parallelepipeds is defined by curves of the R dimension versus the L dimension for various values of the V dimension. The capacity to contain cylinders is defined by curves in which the diameter D is a function of the length H for each of three orientations: H parallel to the R, L, and V axes.

CAVITIES 1-1, 1-2, & 1-3

LOCATION: LEM ADAPTER FAIRING, AROUND APOLLO NOZZLE
EFFECTIVITY: SA-207 AND ON
VOLUME: 163,966 IN^3
WEIGHT CAPACITY: 1,000 LBS



STANDARD SHAPES CAPACITY

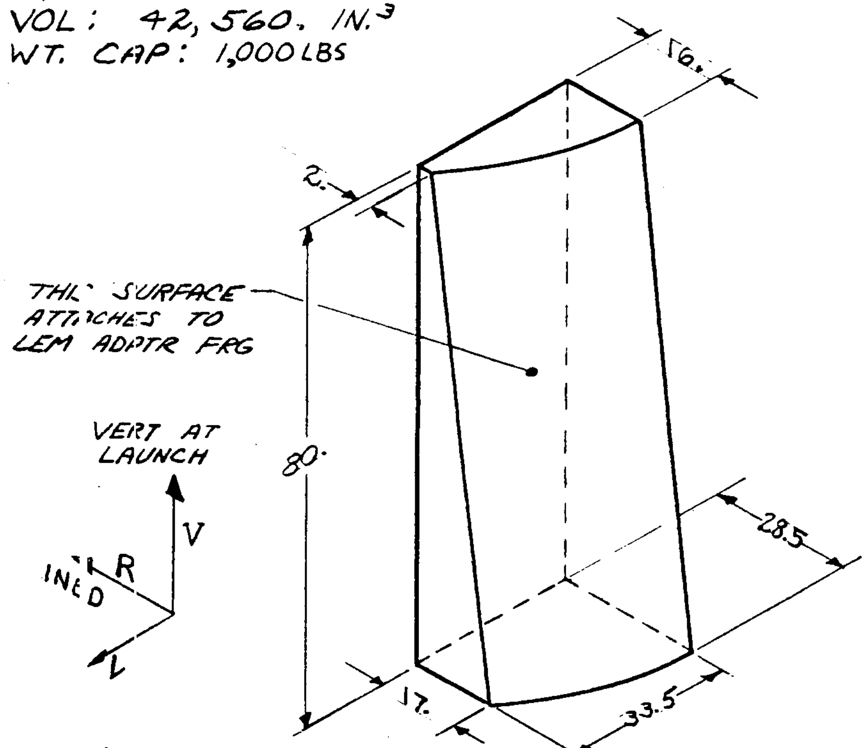
TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE	4	24.5 (DIA)

BROOKS 9-9-65

Figure A-1 DESCRIPTIVE DRAWING - CAVITIES 1-1, 1-2, & 1-3

CAVITIES 2-1 & 2-7

LOCATION: LEM ADAPTER FRG
EFF: SA-207 AND ON
VOL: 42,560. $IN.^3$
WT. CAP: 1,000 LBS

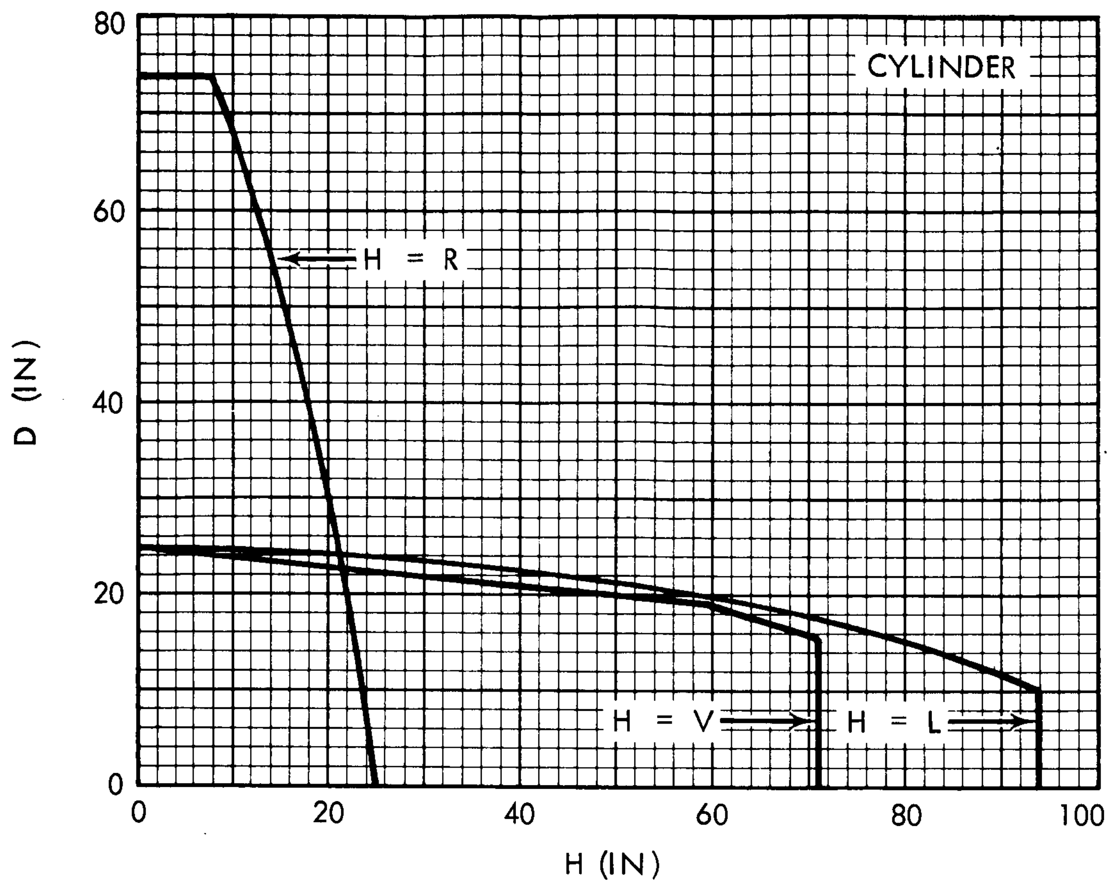


STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE - INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		" " "
SPHERE	1	21. DIA

Barron 8-26-65

Figure A-2 DESCRIPTIVE DRAWING - CAVITIES 2-1 & 2-7



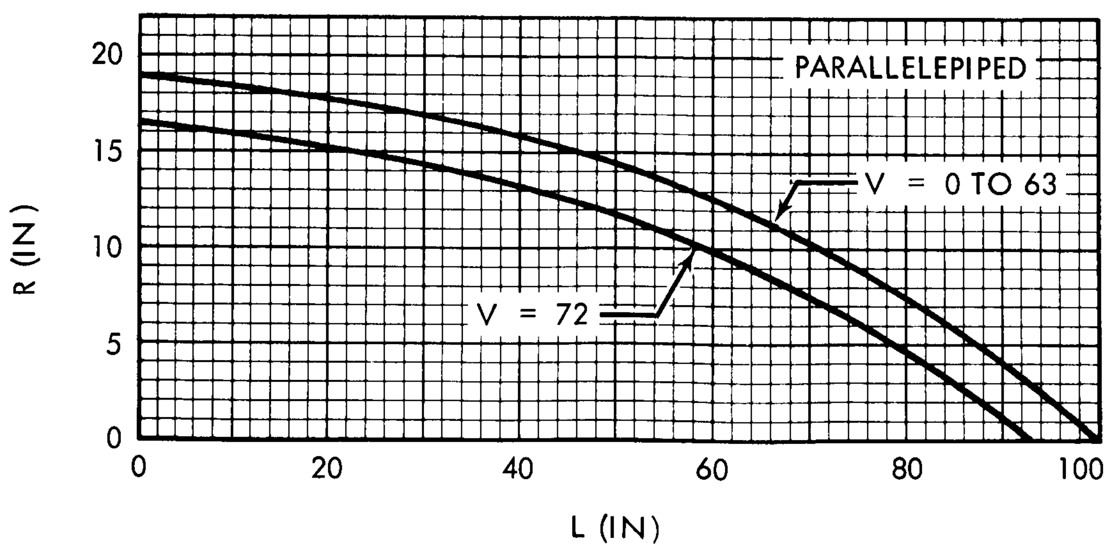
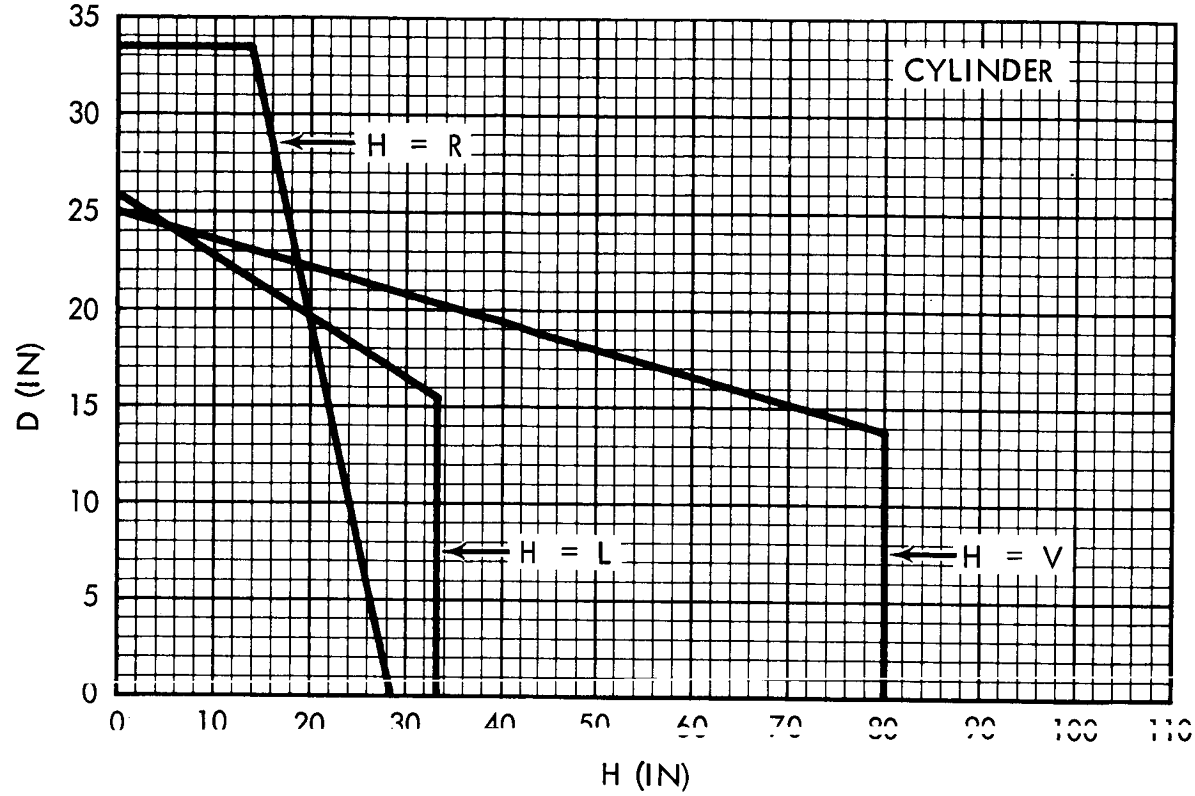


Figure A-3 STANDARD SHAPES CAPACITY - CAVITIES 1-1, 1-2, & 1-3



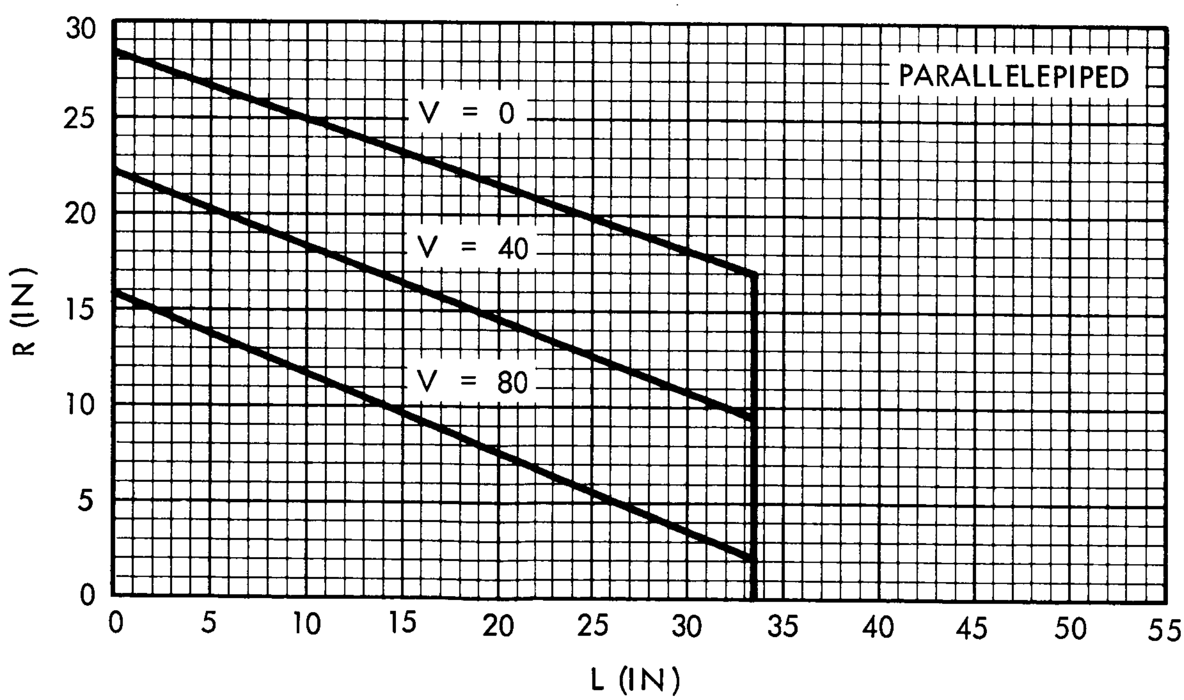
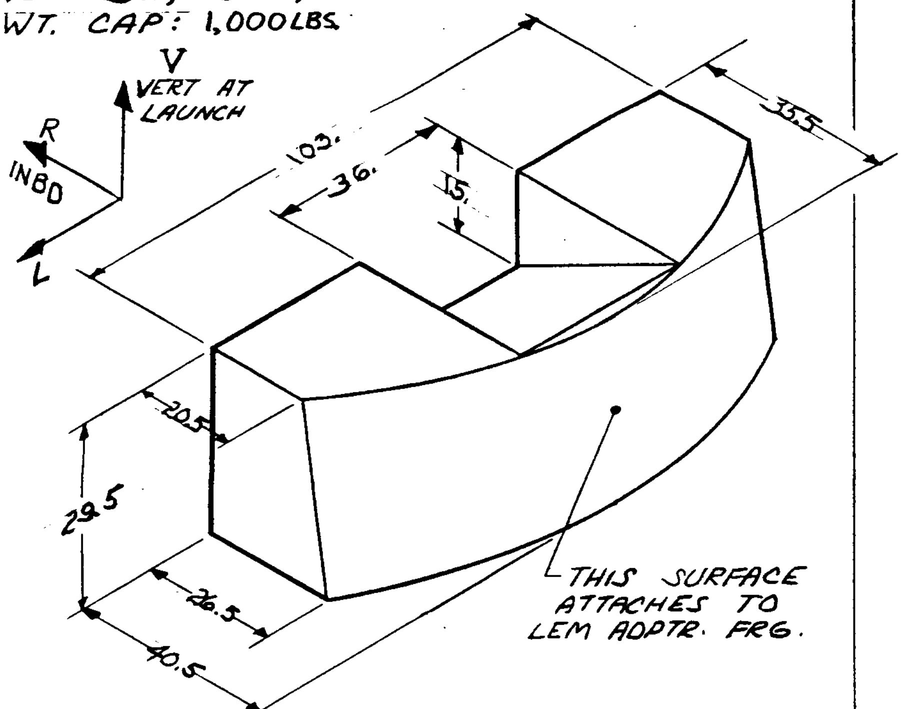


Figure A-4 STANDARD SHAPES CAPACITY - CAVITIES 2-1 & 2-7

CAVITY 2-2

LOCATION: LEM ADAPTER FRG.
EFF: SA-207 AND ON
VOL: 92,030. IN^3
WT. CAP: 1,000 LBS.



STANDARD SHAPES CAPACITY

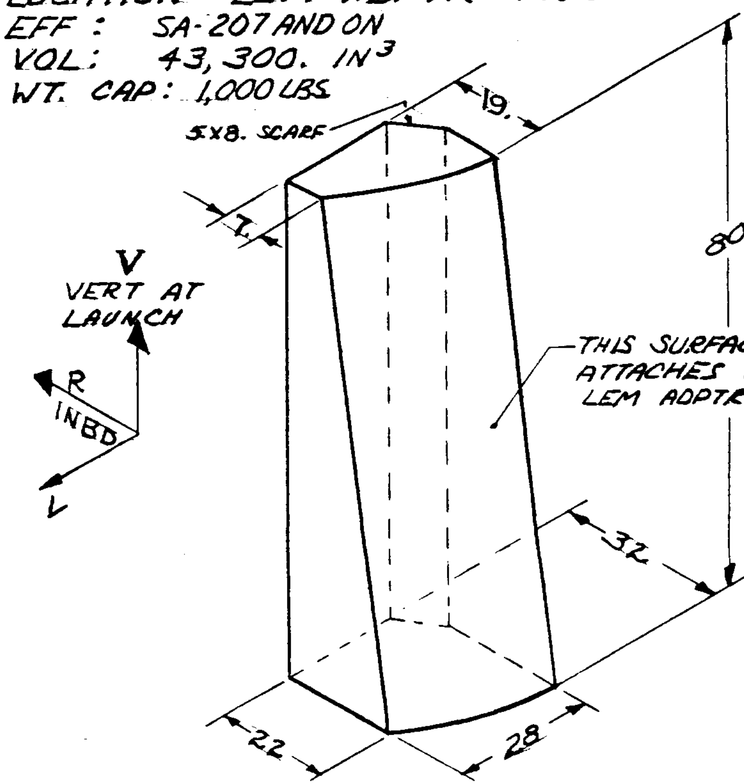
TYPE	NO	SIZE - INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		" " "
SPHERE	2	29.5 DIA

8-26-65

Figure A-5 DESCRIPTIVE DRAWING - CAVITY 2-2

CAVITIES 2-3 & 2-4

LOCATION: LEM ADPTR FRG.
EFF: SA-207 AND ON
VOL: 43,300. IN^3
WT. CAP: 1,000 LBS.



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE - INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		" " "
SPHERE	1	25.0 DIA

8-26-65

Figure A-6 DESCRIPTIVE DRAWING - CAVITIES 2-3 & 2-4

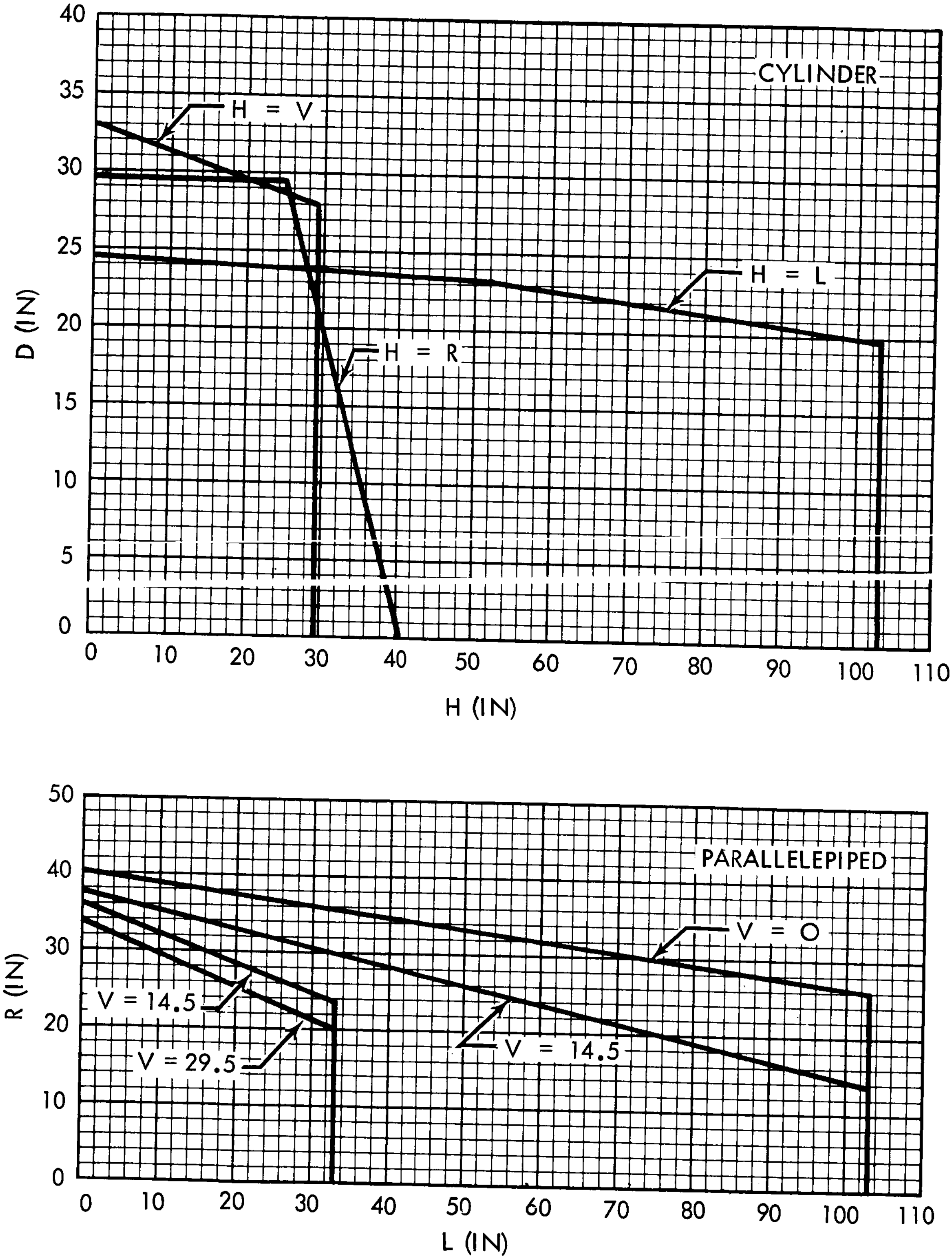


Figure A-7 STANDARD SHAPES CAPACITY - CAVITY 2-2

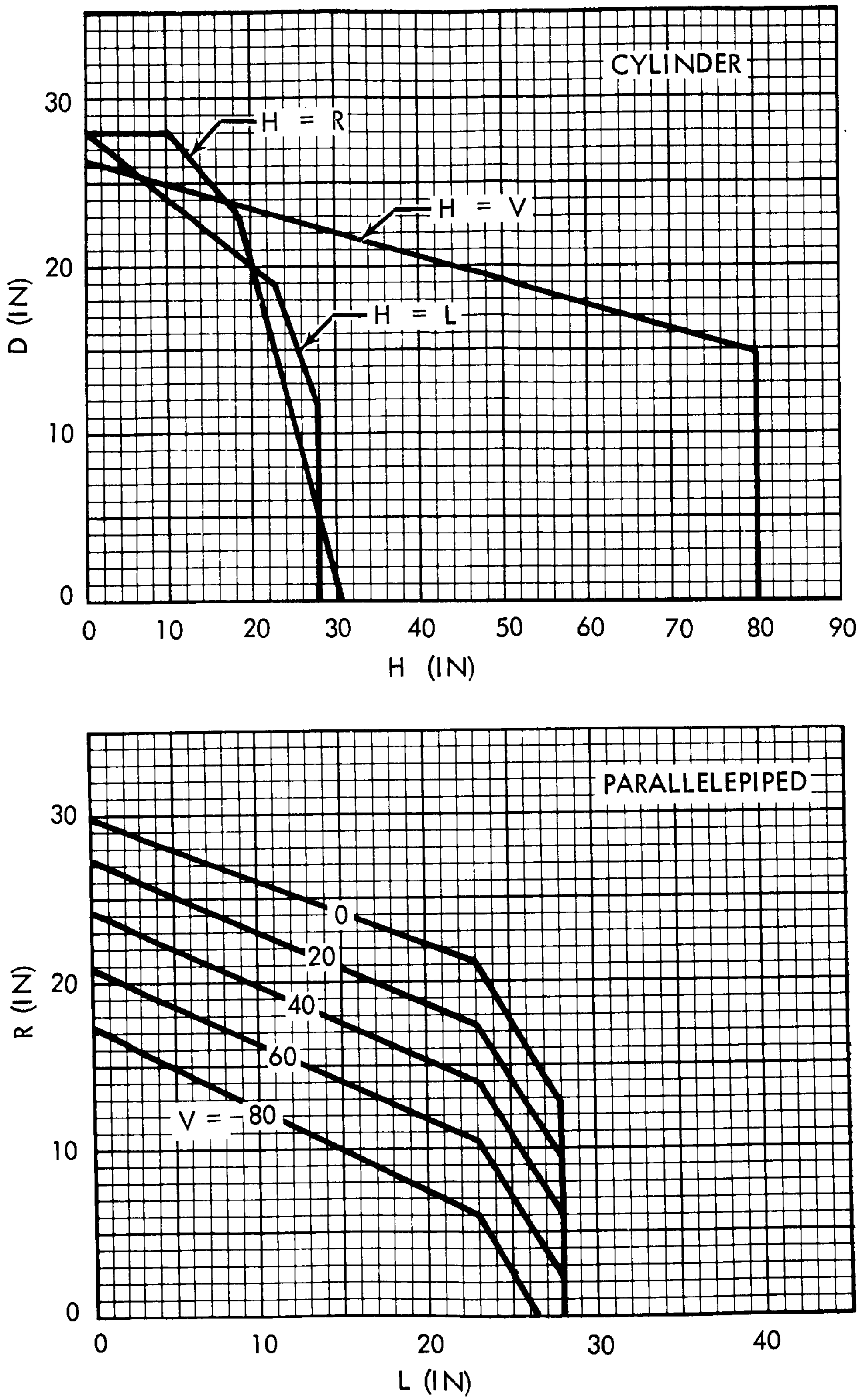
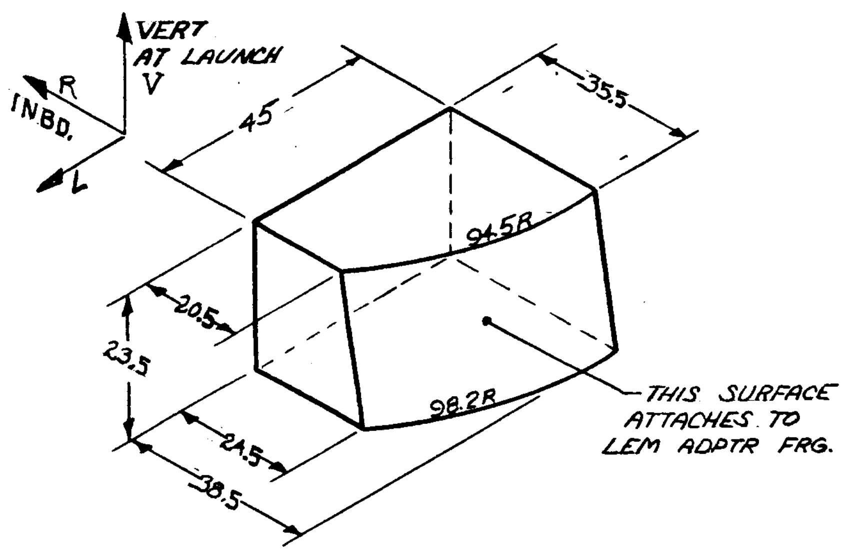


Figure A-8 STANDARD SHAPES CAPACITY - CAVITIES 2-3 & 2-4

CAVITIES 2-5 & 2-6

LOCATION: LEM ADPTR. FRG.
EFF: SA-207 AND ON
VOL: ~30,000. IN.^{3}
WT CAP: 1,000 LBS.



STANDARD SHAPES CAPACITY

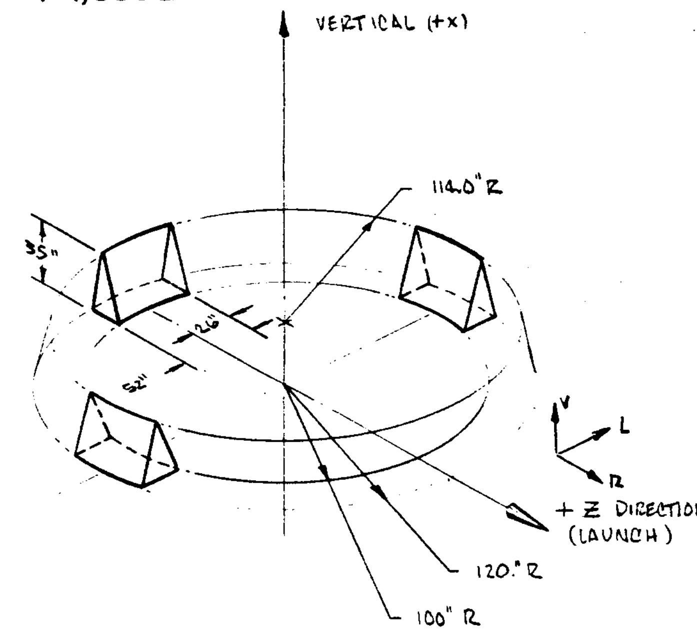
TYPE	NO	SIZE- INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		" " "
SPHERE	1	23.5 DIA.

Baron 8-27-65

Figure A-9 DESCRIPTIVE DRAWING - CAVITIES 2-5 AND 2-6

CAVITIES 3-1, 3-3 & 3-7

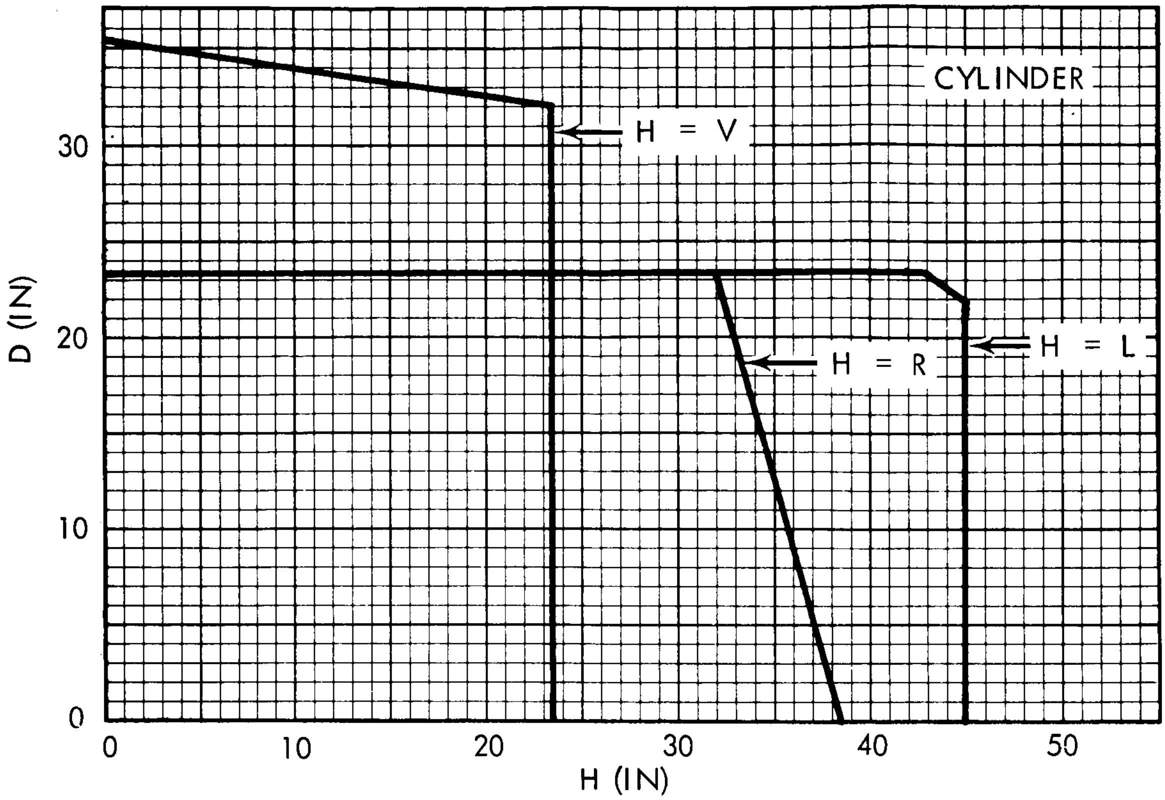
LOCATION: AROUND LEM DESCENT STAGE
EFF: SA-207 AND ON
VOL: 18,200 IN^3
WT. CAP: 1,000 LBS



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE	3	15.0 DIA

Figure A-10 DESCRIPTIVE DRAWING - CAVITIES 3-1, 3-3, & 3-7



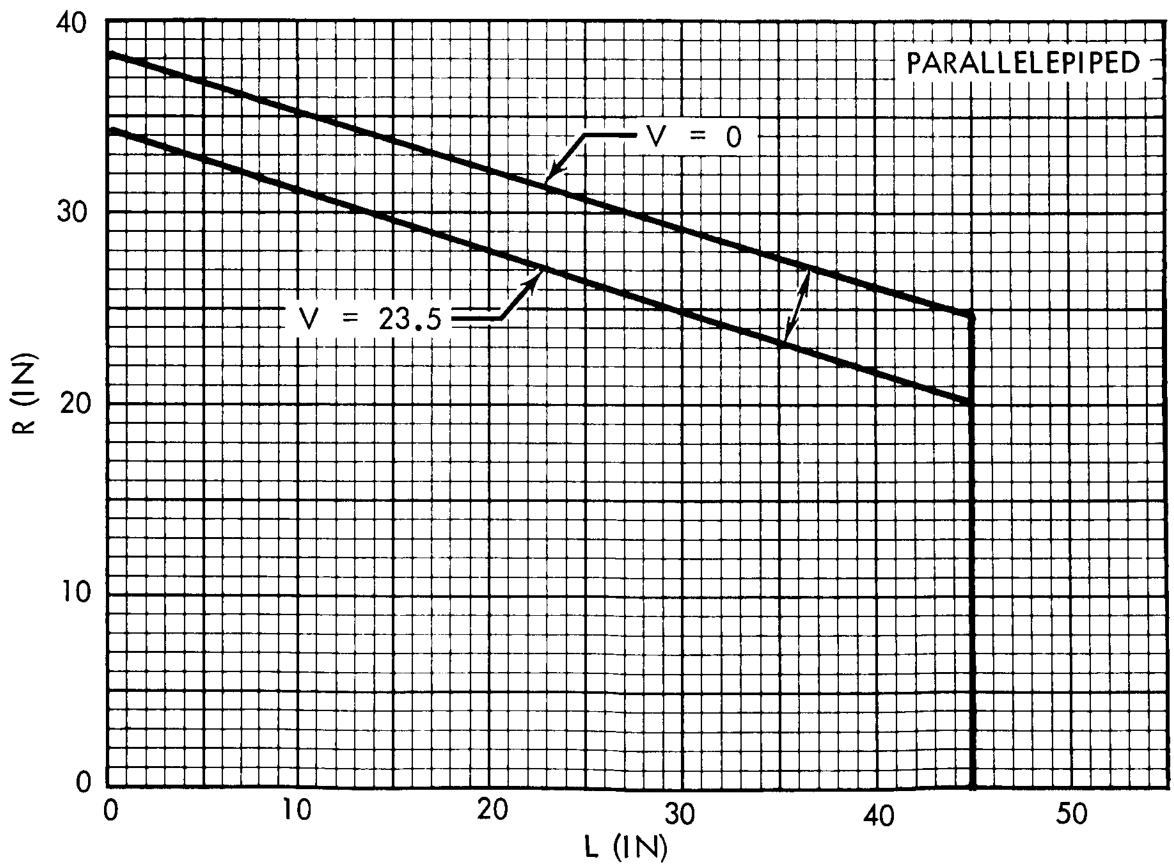
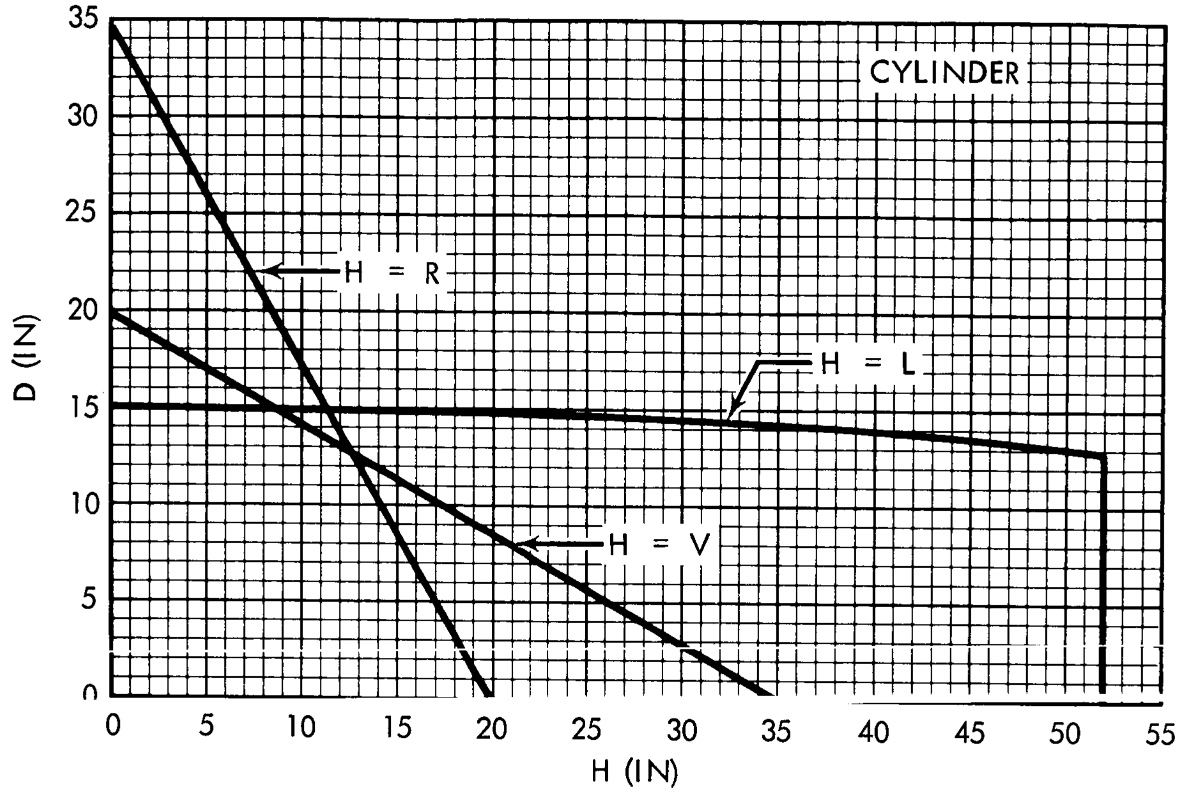


Figure A-11 STANDARD SHAPES CAPACITY - CAVITIES 2-5 & 2-6



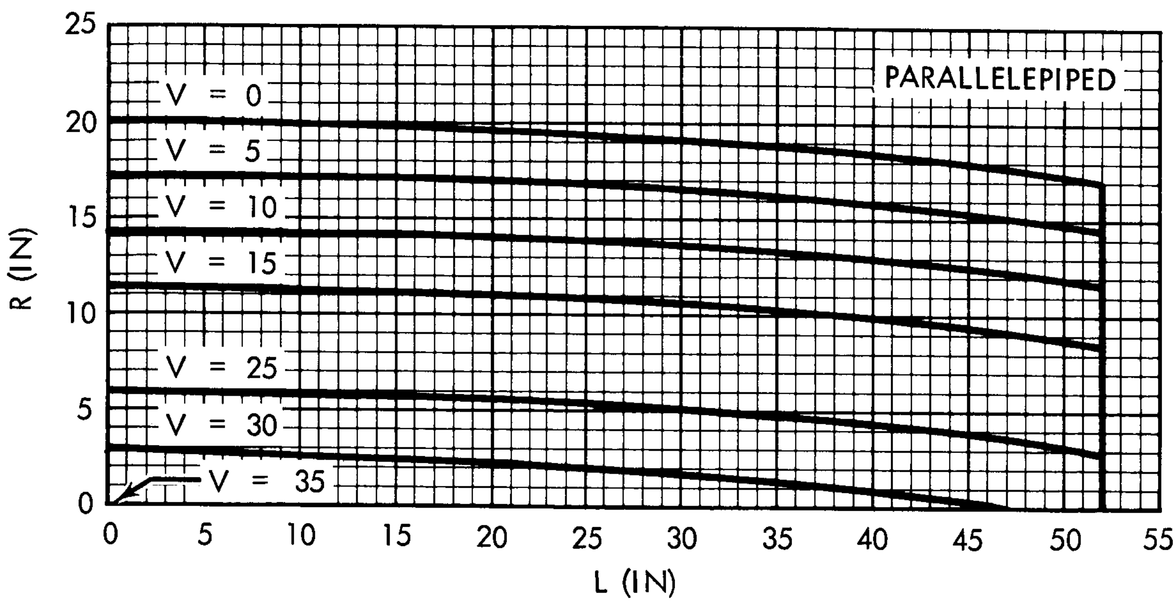
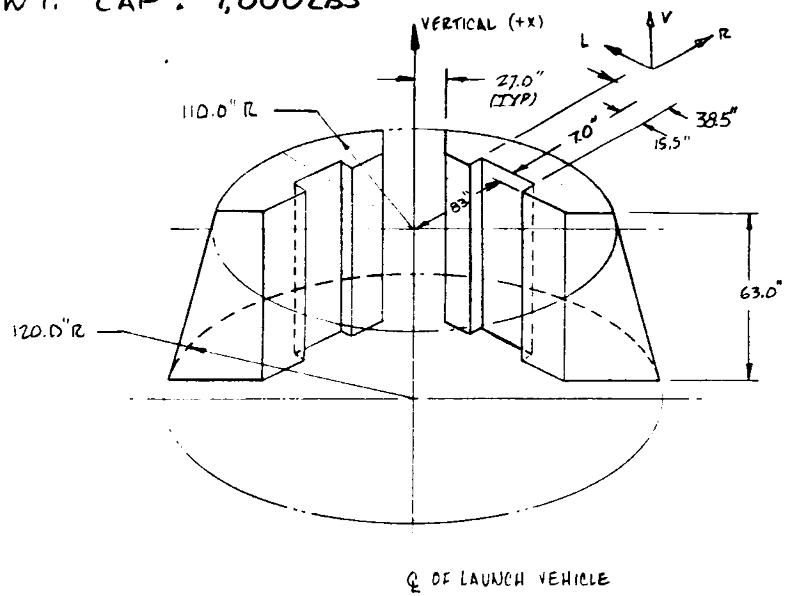


Figure A-12 STANDARD SHAPES CAPACITY - CAVITIES 3-1, 3-3 & 3-7

CAVITIES 3-2, 3-4, & 3-6

LOCATION: AROUND LEM DESCENT STAGE
EFF: SA-207 AND ON
VOL: 171,039 IN^3 EACH
WT. CAP: 1,000 LBS



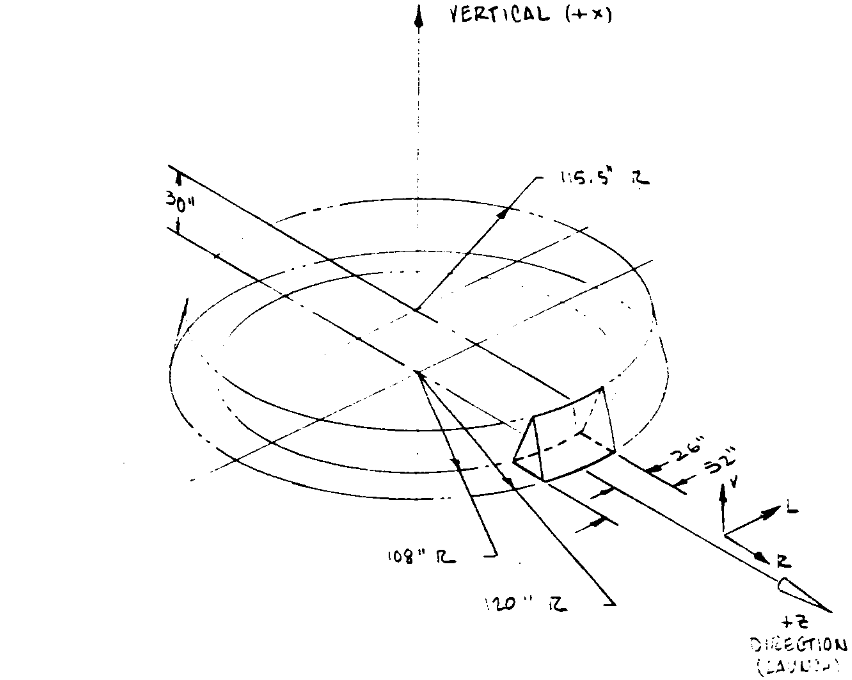
STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEP		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE		28.0 DIA

Figure A-13 DESCRIPTIVE DRAWING - CAVITIES 3-2, 3-4, & 3-6

CAVITY 3-5

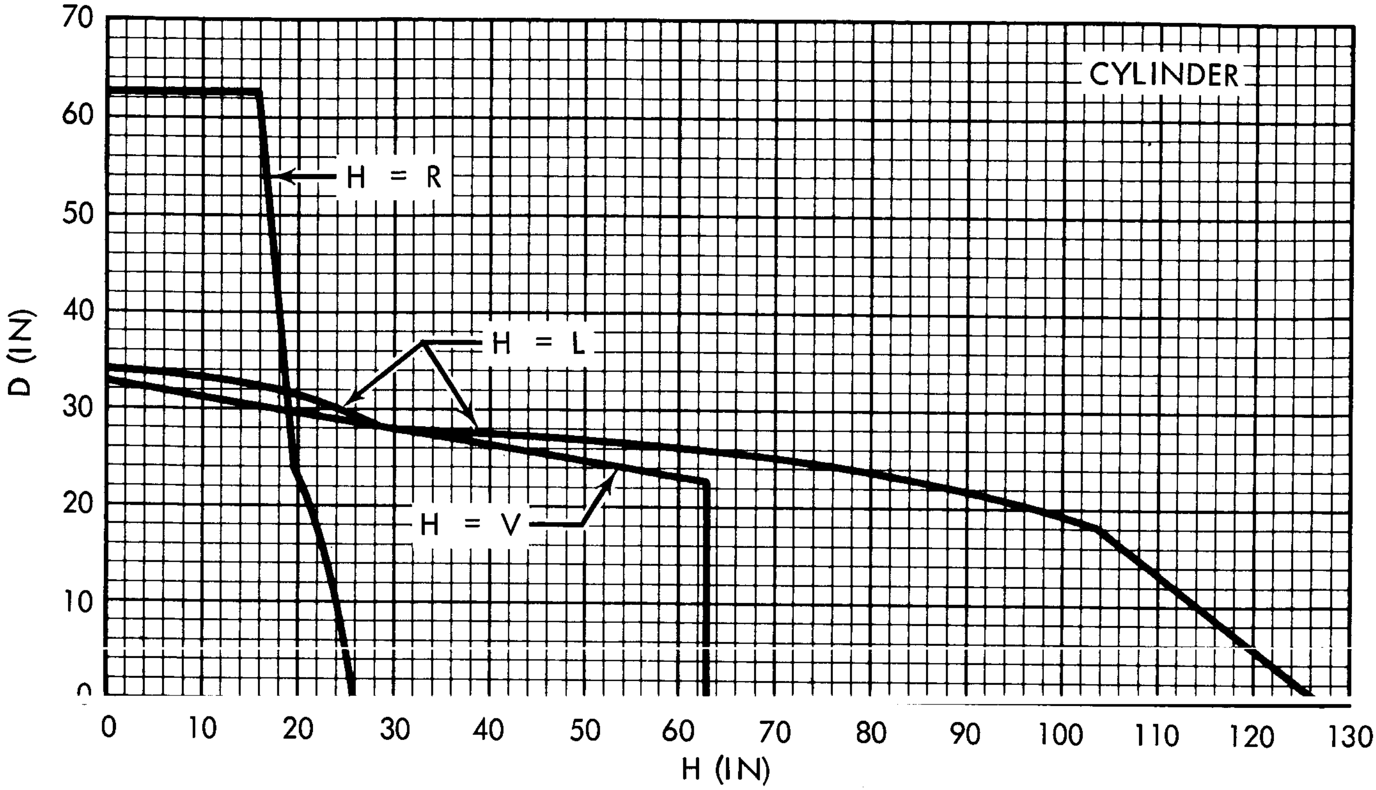
LOCATION: AROUND LEM DESCENT STAGE
EFF: SA-207 AND ON
VOL: 9,350 IN^3
WT. CAP: 1,000 LBS



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEP		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE	1	10.0 DIA

Figure A-14 DESCRIPTIVE DRAWING - CAVITY 3-5



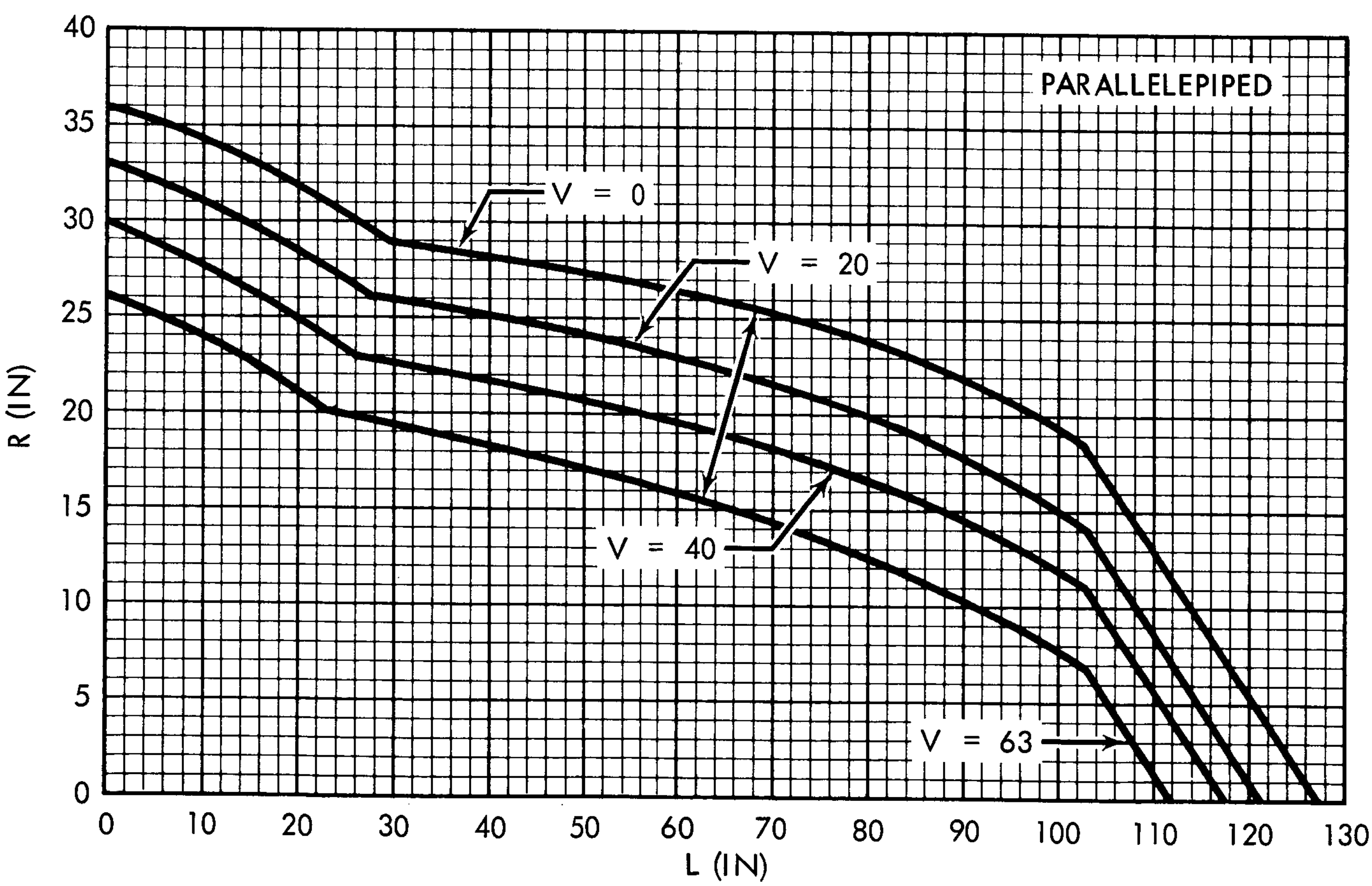


Figure A-15 STANDARD SHAPES CAPACITY - CAVITIES 3-2, 3-4, & 3-6

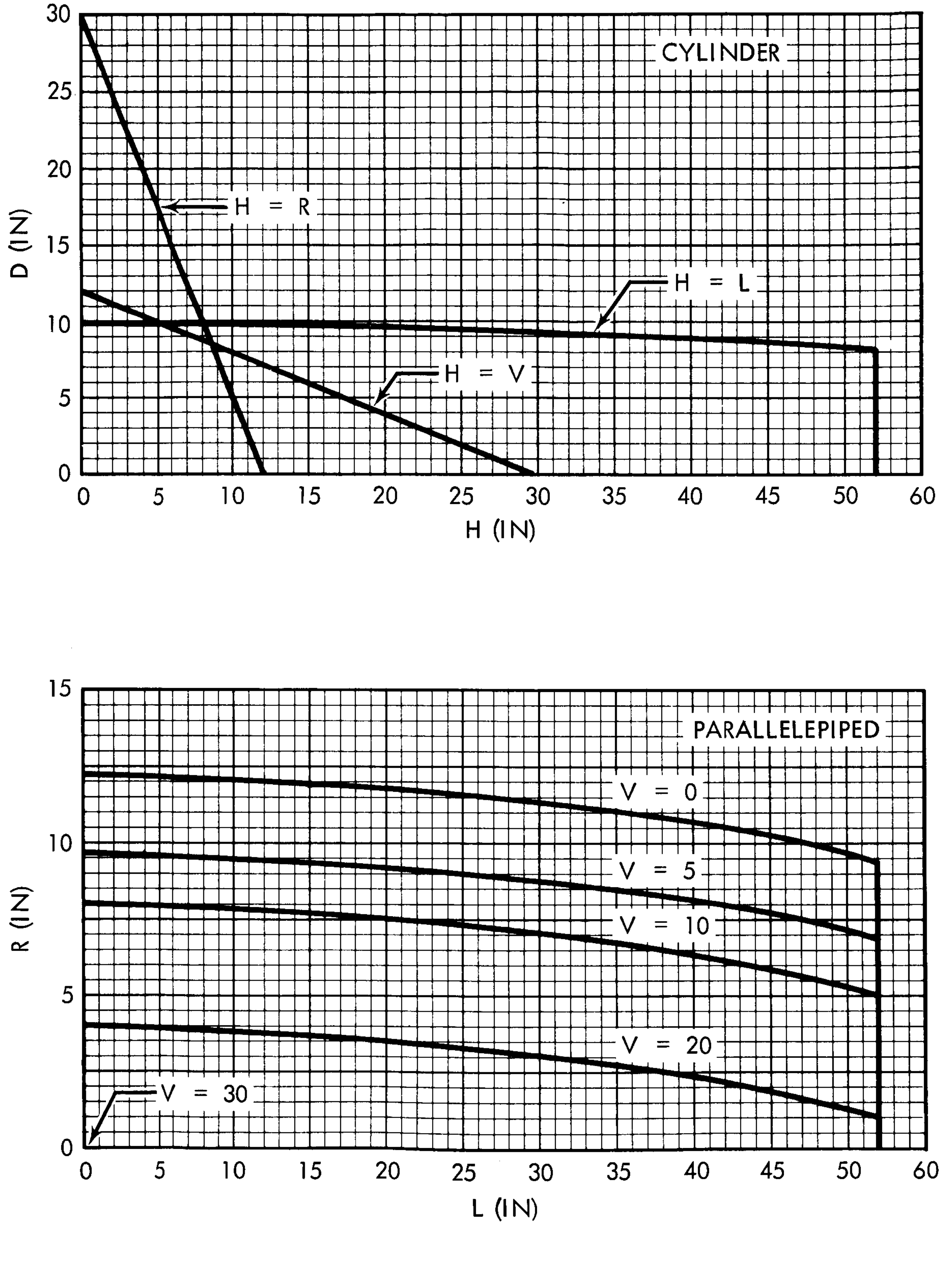
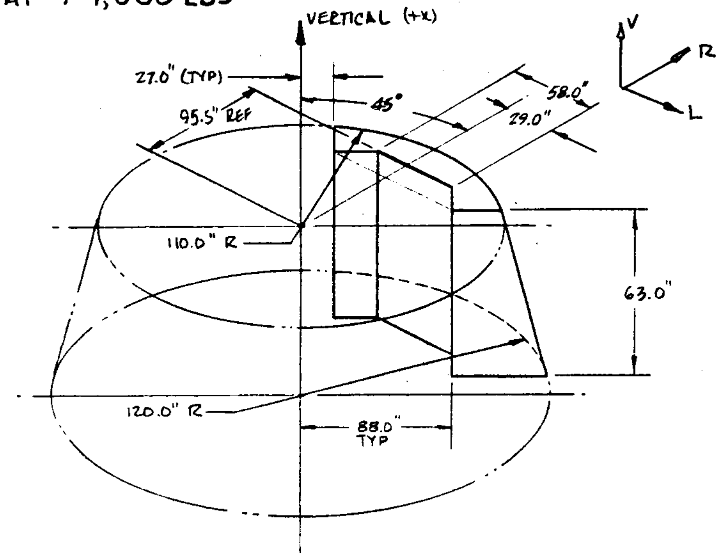


Figure A-16 STANDARD SHAPES CAPACITY - CAVITY 3-5

CAVITY 3-8

LOCATION: AROUND LEM DESCENT STAGE
EFF: SA-207 AND ON
VOL: 132,500 IN³
WT. CAP: 1,000 LBS



VIEW LOOKING AFT ~ AS IF SEATED IN LEM ~ LAUNCH POSITION
CAVITY IN AFT QUADRANT / RHS (LOOKING AFT)
BETWEEN LEM LANDING GEAR SUPPORTS

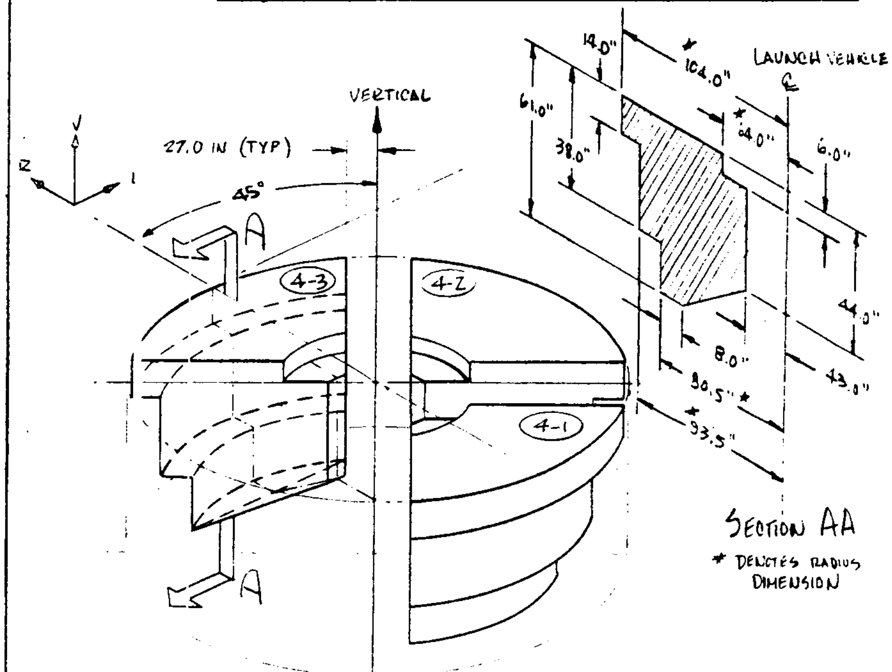
STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEPIPED		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE	1	225 DIA

Figure A-17 DESCRIPTIVE DRAWING - CAVITY 3-8

CAVITIES 4-1, 4-2, 4-3

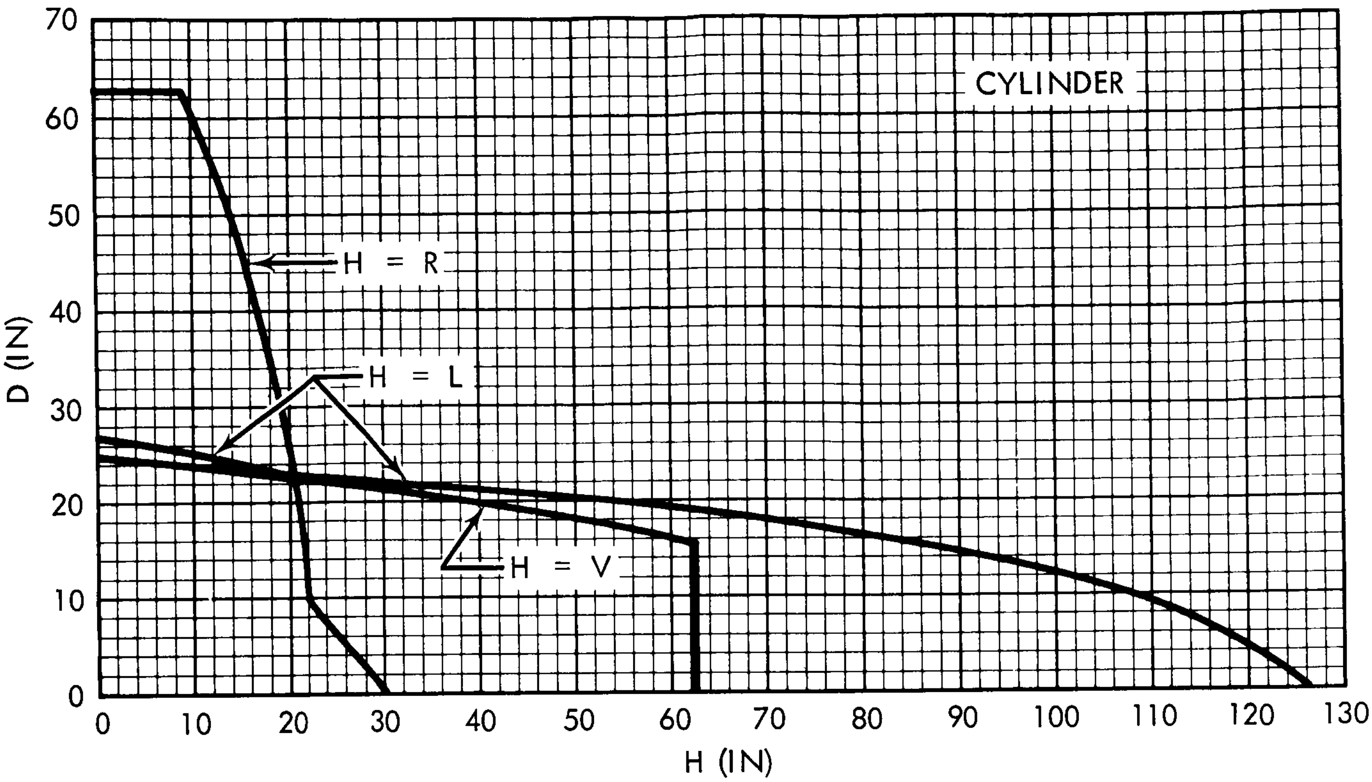
LOCATION: BETWEEN LEM LANDING LEGS
EFF: SA-207 AND ON
VOL: 138,080 IN³ each
WT. CAP: 2,500 LBS



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET.
SPHERE	1	385 DIA

Figure A-18 DESCRIPTIVE DRAWING - CAVITIES 4-1, 4-2, & 4-3



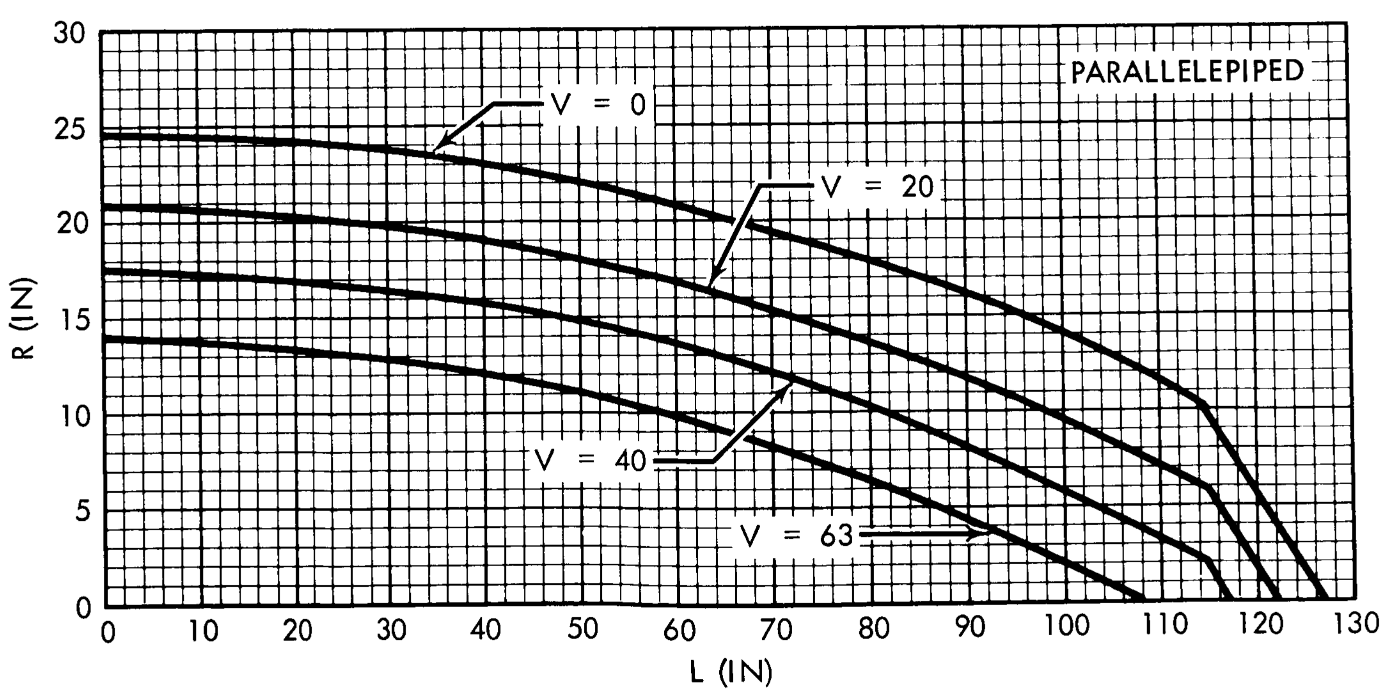
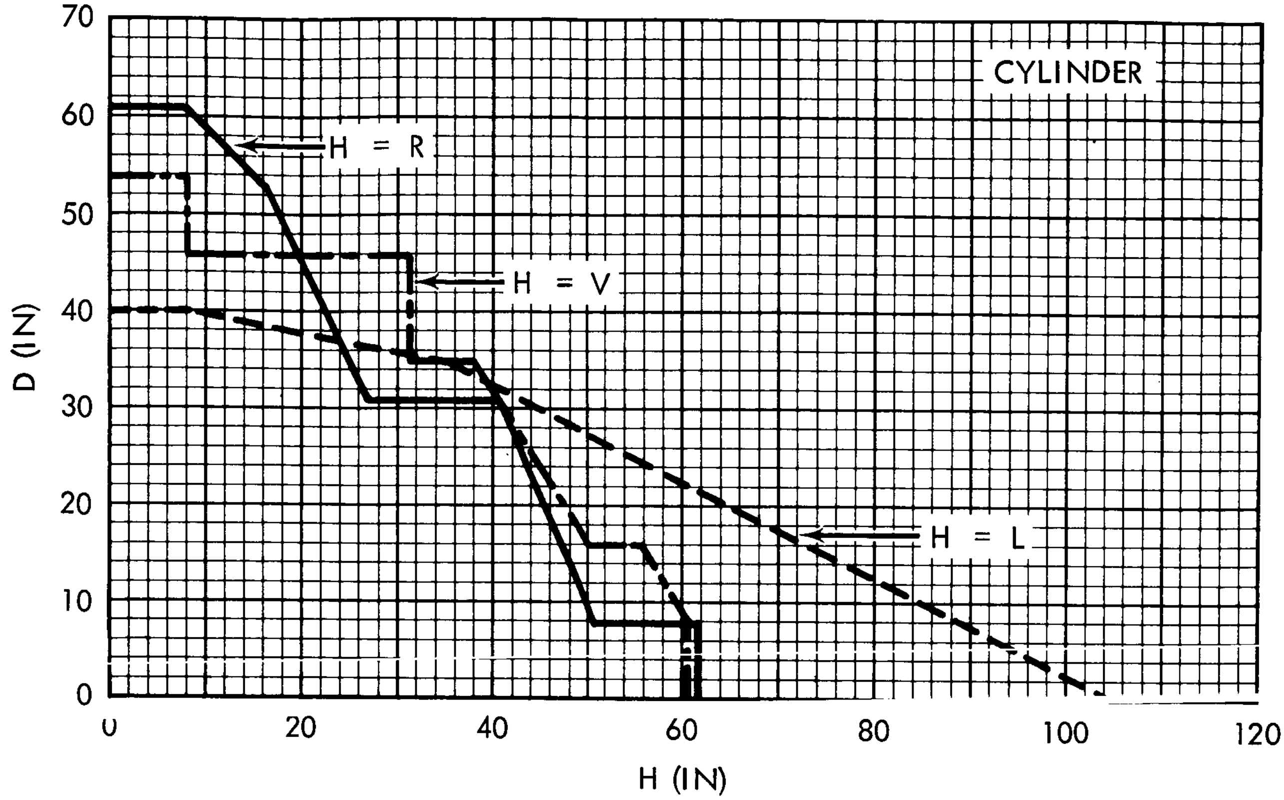


Figure A-19 STANDARD SHAPES CAPACITY - CAVITY 3-8



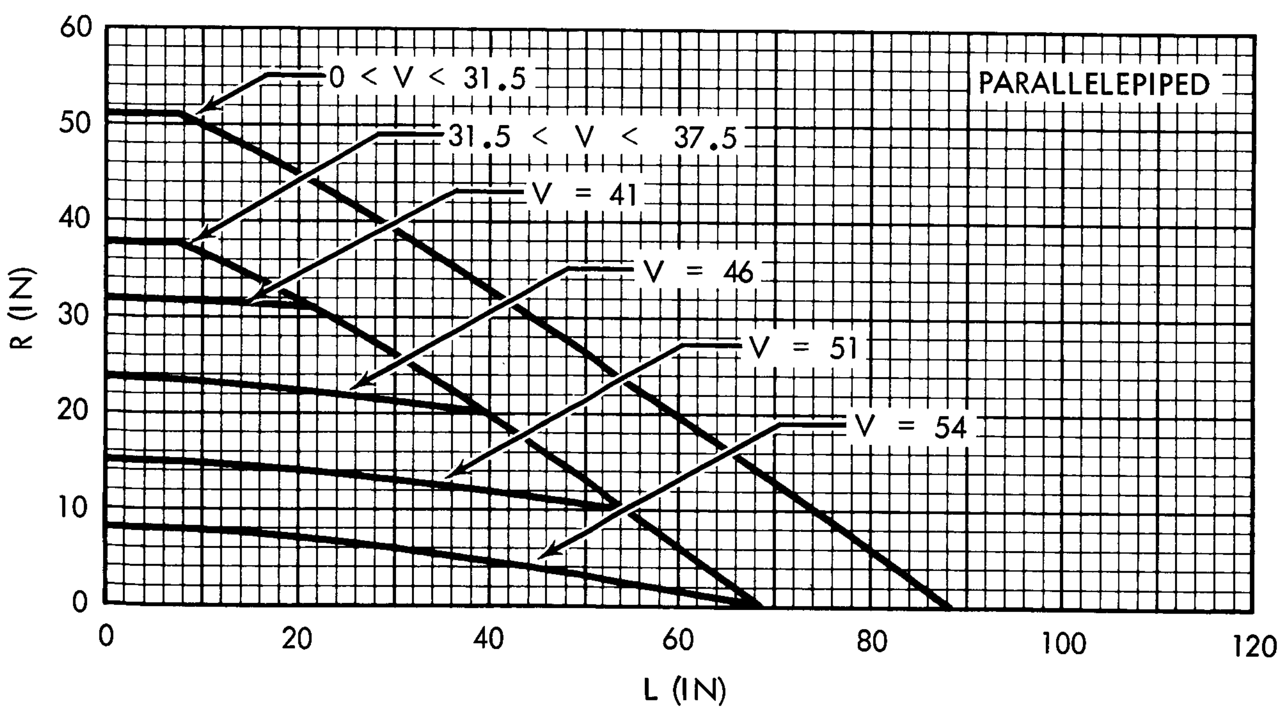
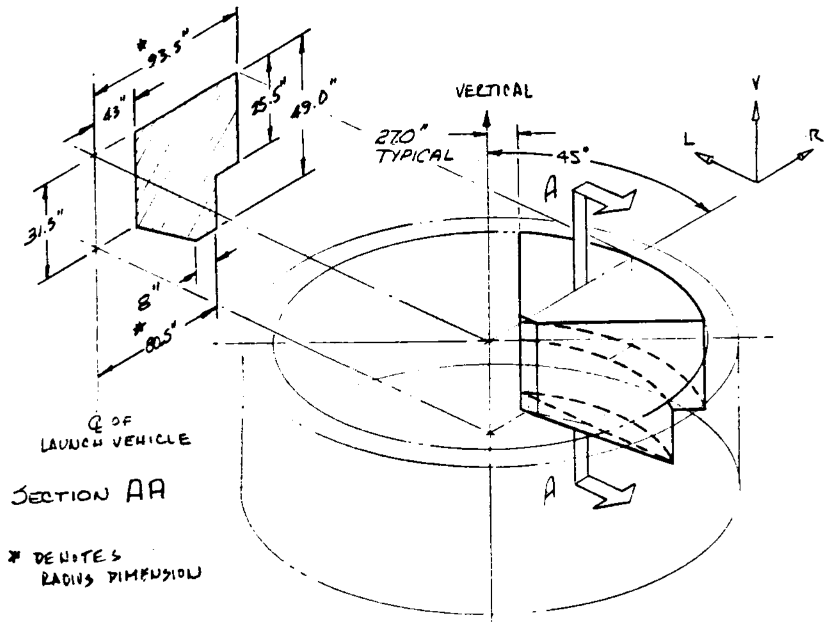


Figure A-20 STANDARD SHAPES CAPACITY - CAVITIES 4-1, 4-2, & 4-3

CAVITY 4-4

LOCATION: AROUND LEM DESCENT STAGE
EFF: SA-207 AND ON
VOL: 94,280 IN³
WT. CAP.: 2,500 LBS



VIEW LOOKING AFT ~ AS IF SEATED IN LEM - LAUNCH POSITION
CAVITY IN AFT QUADRANT / RHS (LOOKING AFT)
BETWEEN STOWED LEM LANDING LEGS

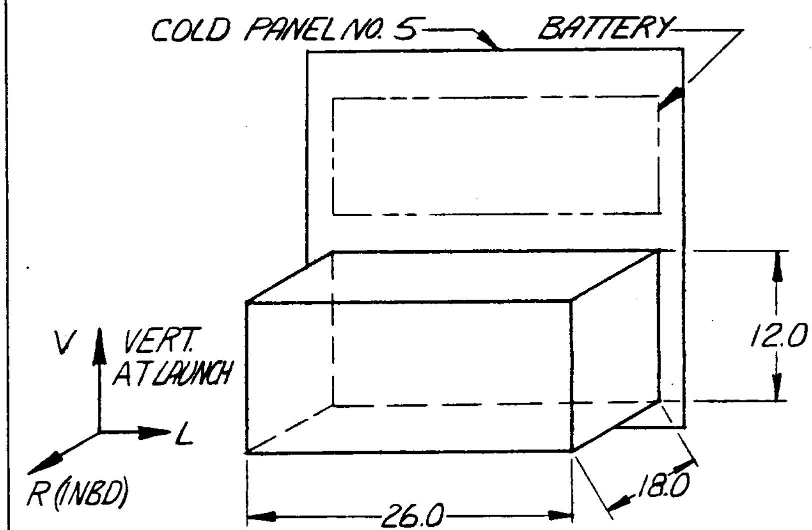
STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE	1	38.5 DIA

Figure A-21 DESCRIPTIVE DRAWING - CAVITY 4-4

CAVITY 5-1

LOCATION: INSTRUMENT UNIT COLD PANEL NO. 5
EFFECTIVITY: SA-205 AND ON
VOLUME: 5,616 IN³
WT. CAPACITY: 150 LBS

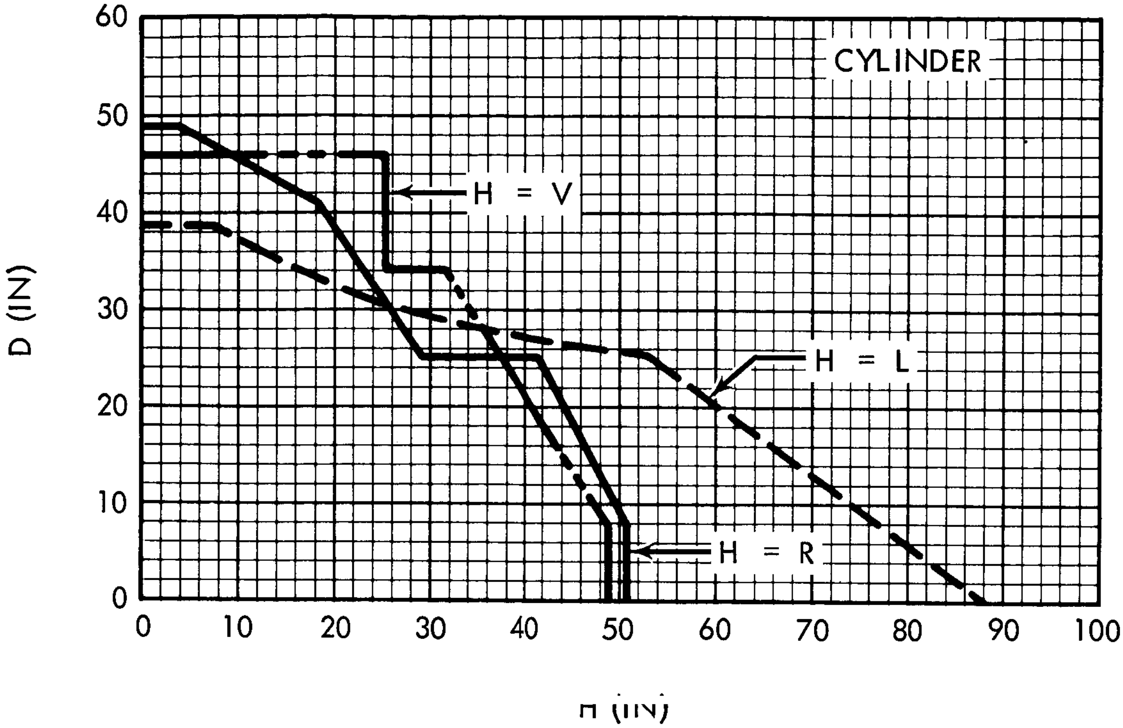


STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	12.0(V) x 18.0(R) x 26.0(L)
CYLINDER (H=V)	1	18.0 (DIA) x 12.0 (H)
CYLINDER (H=R)	1	12.0 (DIA) x 18.0 (H)
CYLINDER (H=L)	1	12.0 (DIA) x 26.0 (H)
SPHERE	1	12.0 (DIA)

BROOKS 9-9-65

Figure A-22 DESCRIPTIVE DRAWING - CAVITY 5-1



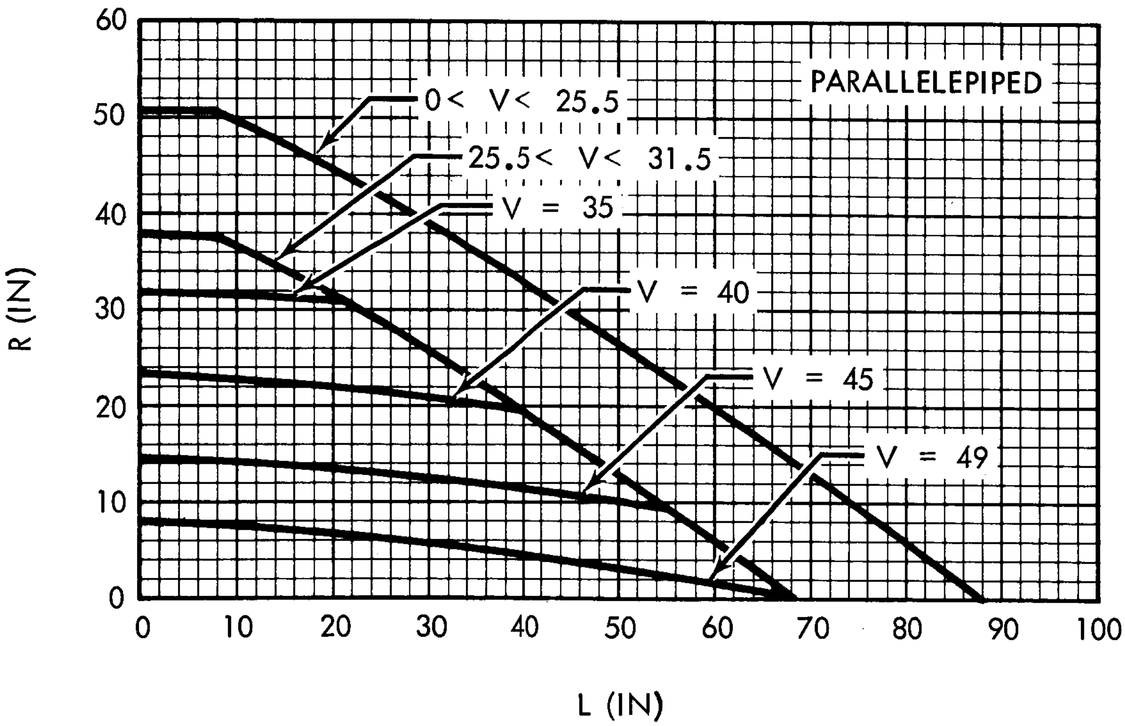
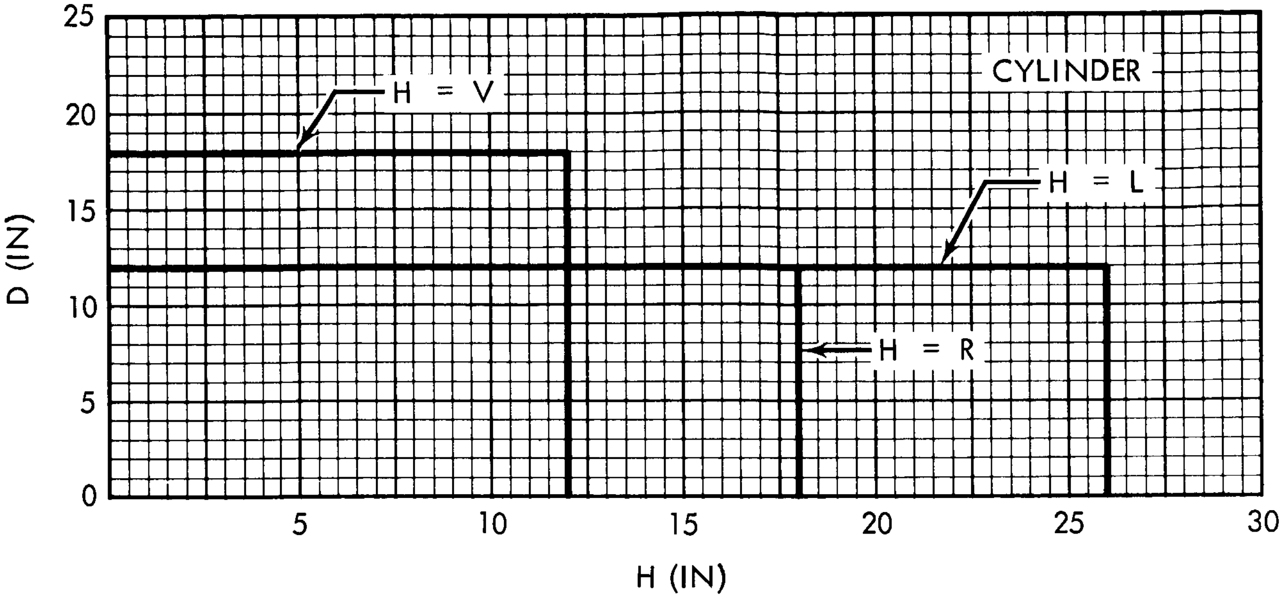


Figure A-23 STANDARD SHAPES CAPACITY - CAVITY 4-4



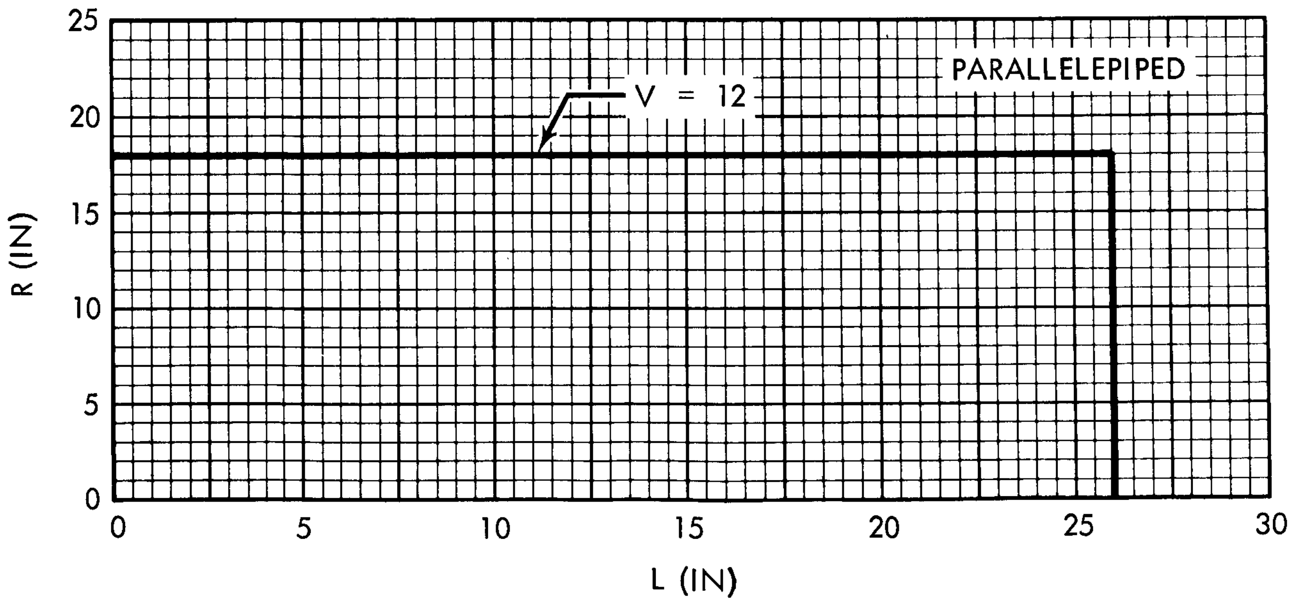
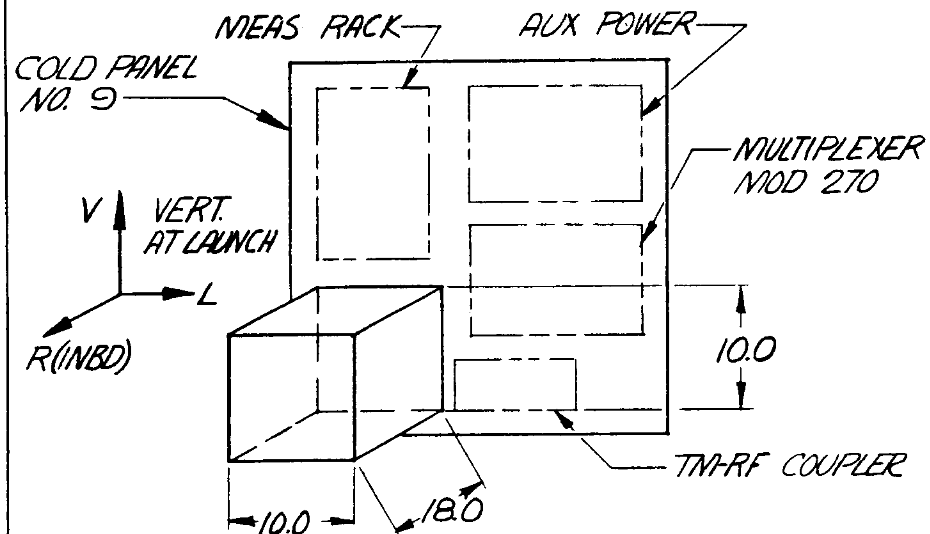


Figure A-24 STANDARD SHAPES CAPACITY - CAVITY 5-1

CAVITY 5-2

LOCATION: INSTRUMENT UNIT COLD PANEL NO. 9
EFFECTIVITY: SA-205 AND ON
VOLUME: 1,800 IN^3
WT. CAPACITY: 276 LBS



STANDARD SHAPES CAPACITY

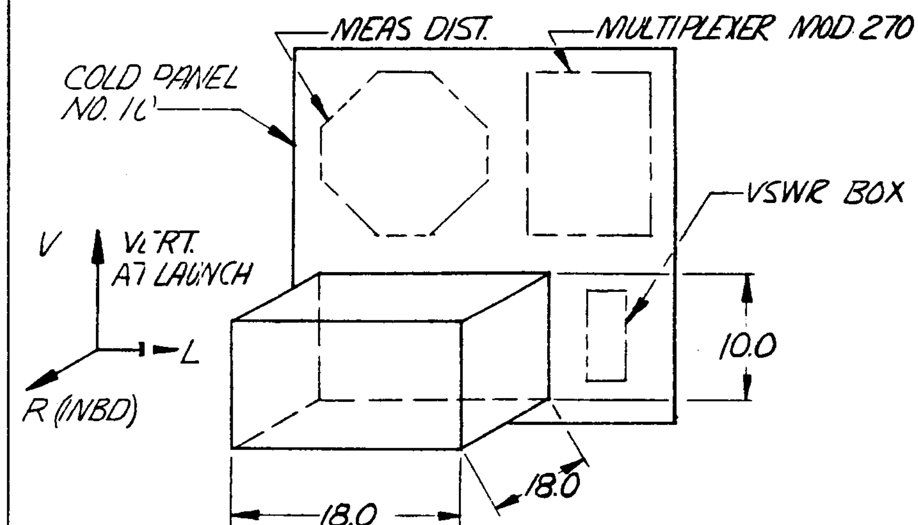
TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	10.0(V) x 18.0(R) x 10.0(L)
CYLINDER (H=V&L)	1	10.0 (DIA) x 10.0(H)
CYLINDER (H=R)	1	10.0 (DIA) x 18.0 (H)
SPHERE	1	10.0 (DIA)

BROOKS 9-9-65

Figure A-25 DESCRIPTIVE DRAWING - CAVITY 5-2

CAVITY 5-3

LOCATION: INSTRUMENT UNIT COLD PANEL NO. 10
EFFECTIVITY: SA-205 AND ON
VOLUME: 3,240 IN^3
WT. CAPACITY: 280 LBS.

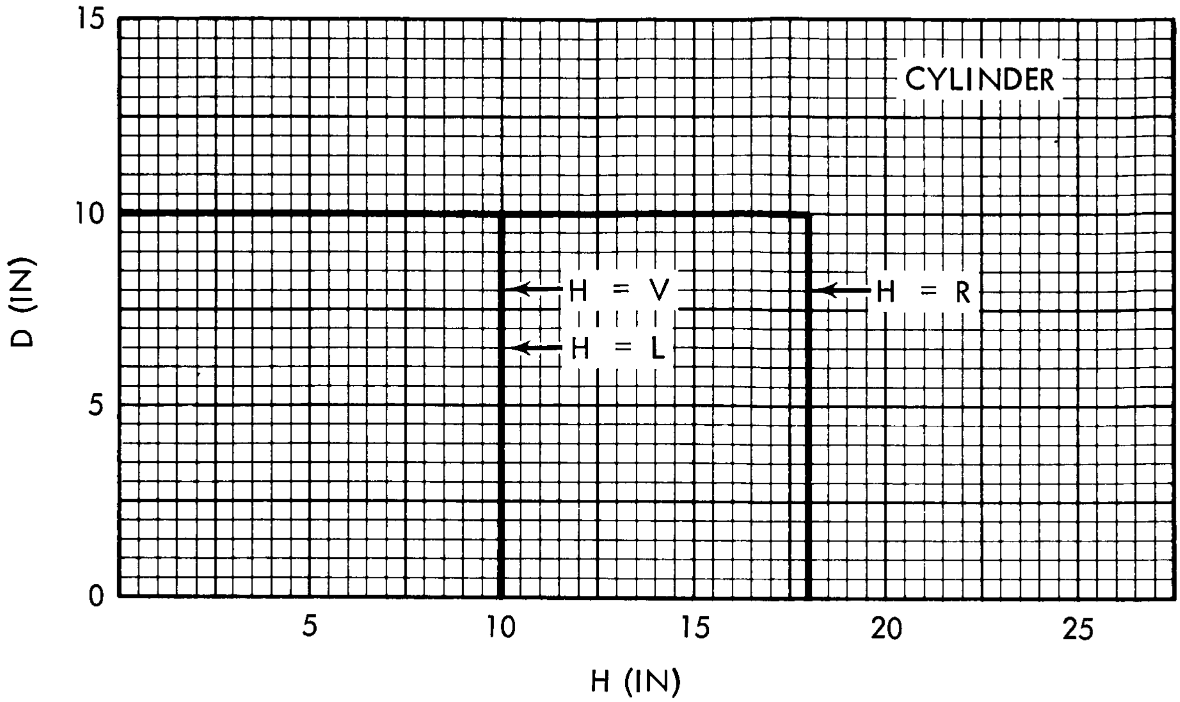


STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	10.0(V) x 18.0(R) x 18.0(L)
CYLINDER (H=V)	1	18.0 (DIA) x 10.0 (H)
CYLINDER (H=R&L)	1	10.0 (DIA) x 18.0 (H)
SPHERE	1	10.0 (DIA)

BROOKS 9-9-65

Figure A-26 DESCRIPTIVE DRAWING - CAVITY 5-3



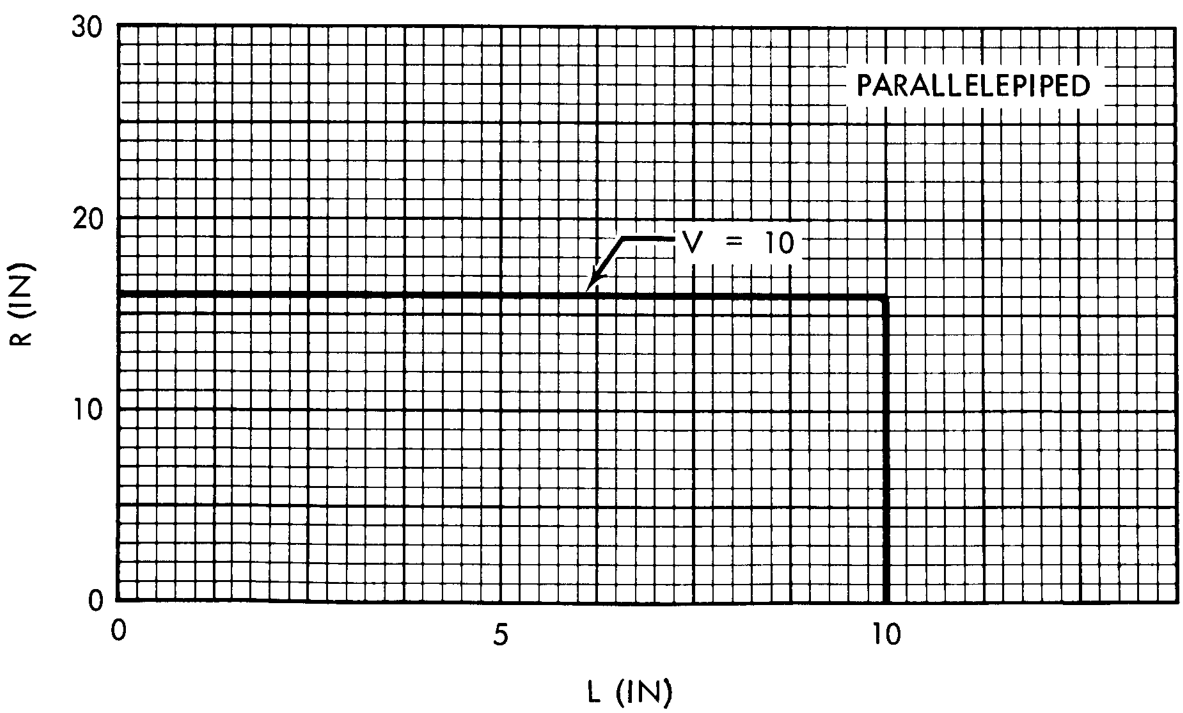
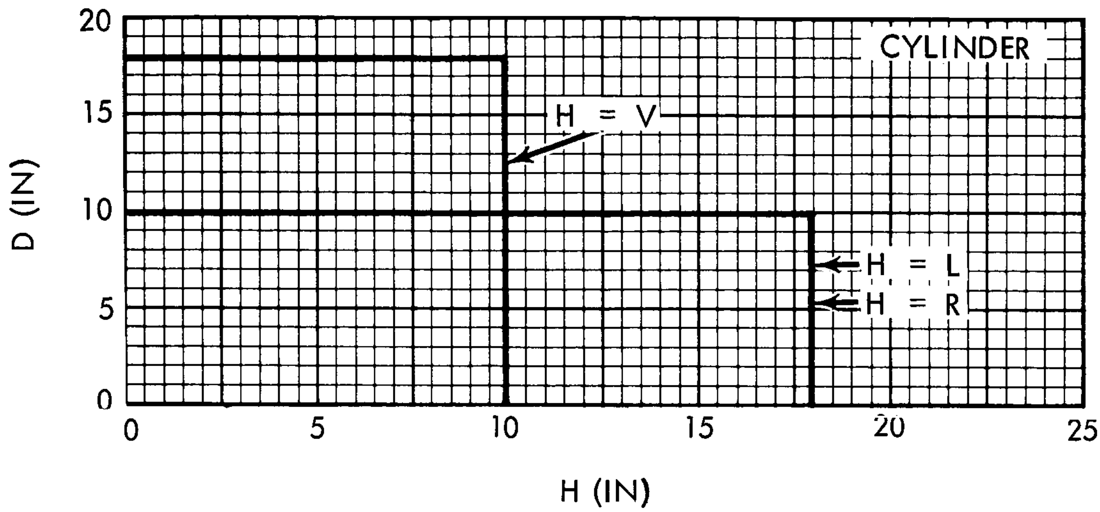


Figure A-27 STANDARD SHAPES CAPACITY - CAVITY 5-2



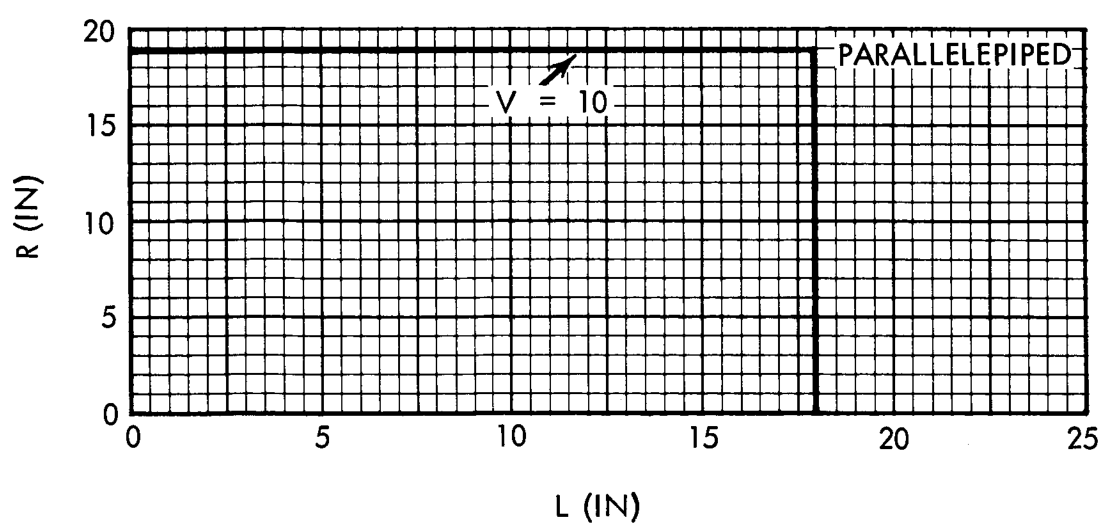
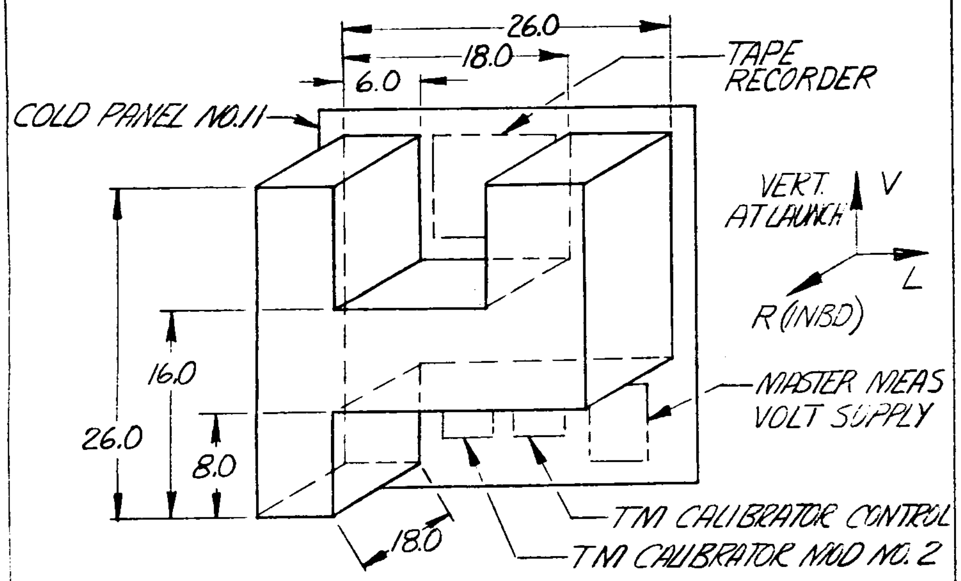


Figure A-28 STANDARD SHAPES CAPACITY - CAVITY 5-3

CAVITY 5-4

LOCATION: INSTRUMENT UNIT COLD PANEL NO. 11
EFFECTIVITY: SA-209 AND ON
VOLUME: 6,950 IN^3
WT. CAPACITY: 208 LBS



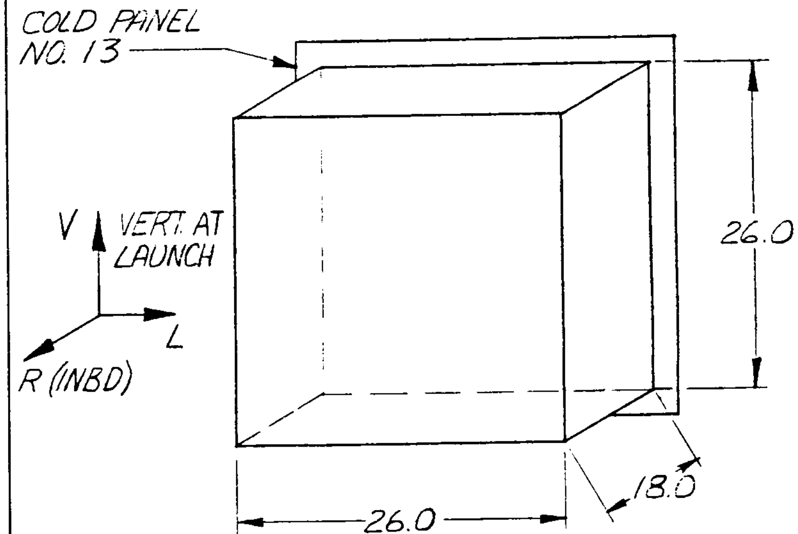
STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	26.0(V) x 18.0(R) x 6.0(L)
PARALLELEPIPED	1	18.0 (V) x 18.0(R) x 8.0(L)
PARALLELEPIPED	1	8.0(V) x 18.0 (R) x 26.0(L)
CYLINDER (H=V)	1	8.0 (DIA) x 18.0(H)
CYLINDER (H=V)	1	6.0 (DIA) x 26.0(H)
CYLINDER (H=V&L)	1	18.0 (DIA) x 8.0(H)
CYLINDER (H=R)	1	9.3 (DIA) x 18.0(H)
CYLINDER (H=L)	1	8.0 (DIA) x 26.0 (H)
SPHERE	1	9.3 (DIA)

Figure A-29 DESCRIPTIVE DRAWING - CAVITY 5-4

CAVITY 5-5

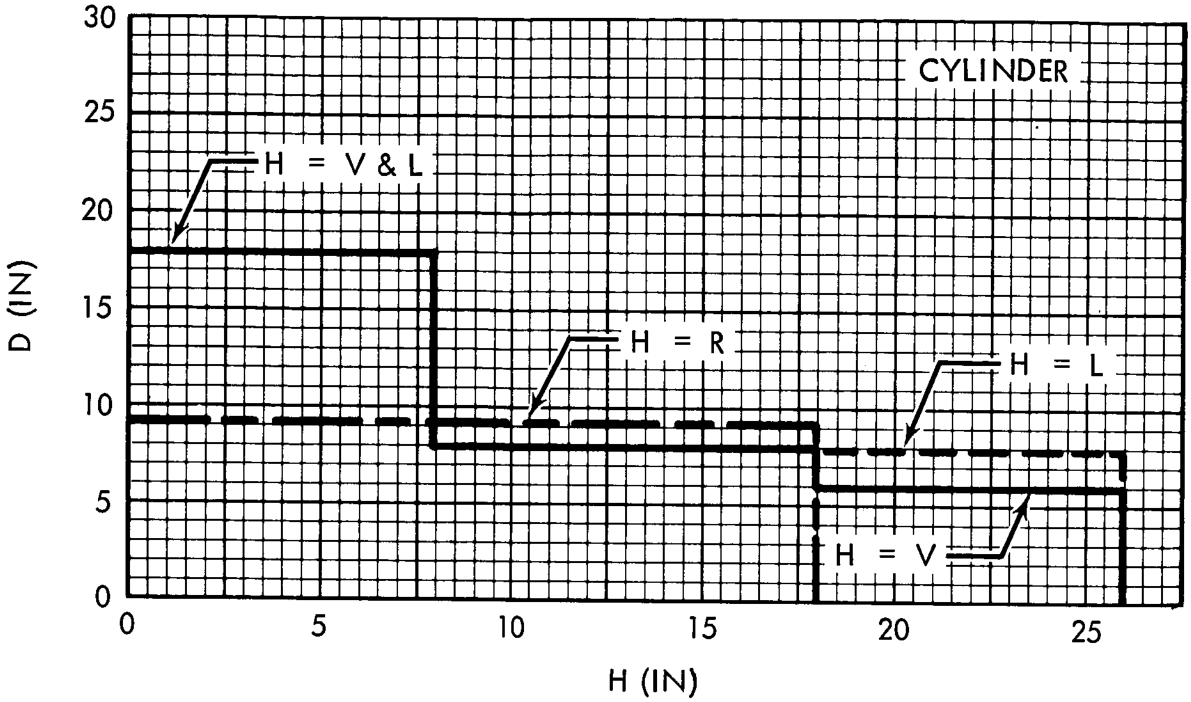
LOCATION: INSTRUMENT UNIT COLD PANEL NO. 13
EFFECTIVITY: SA-205 AND ON
VOLUME: 12,186 IN^3
WT. CAPACITY: 330 LBS.



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	26.0 (V) x 18.0(R) x 26.0(L)
CYLINDER (H=V&L)	1	18.0(DIA) x 26.0 (H)
CYLINDER (H=R)	1	26.0 (DIA) x 18.0 (H)
SPHERE	1	18.0(DIA)

Figure A-30 DESCRIPTIVE DRAWING - CAVITY 5-5



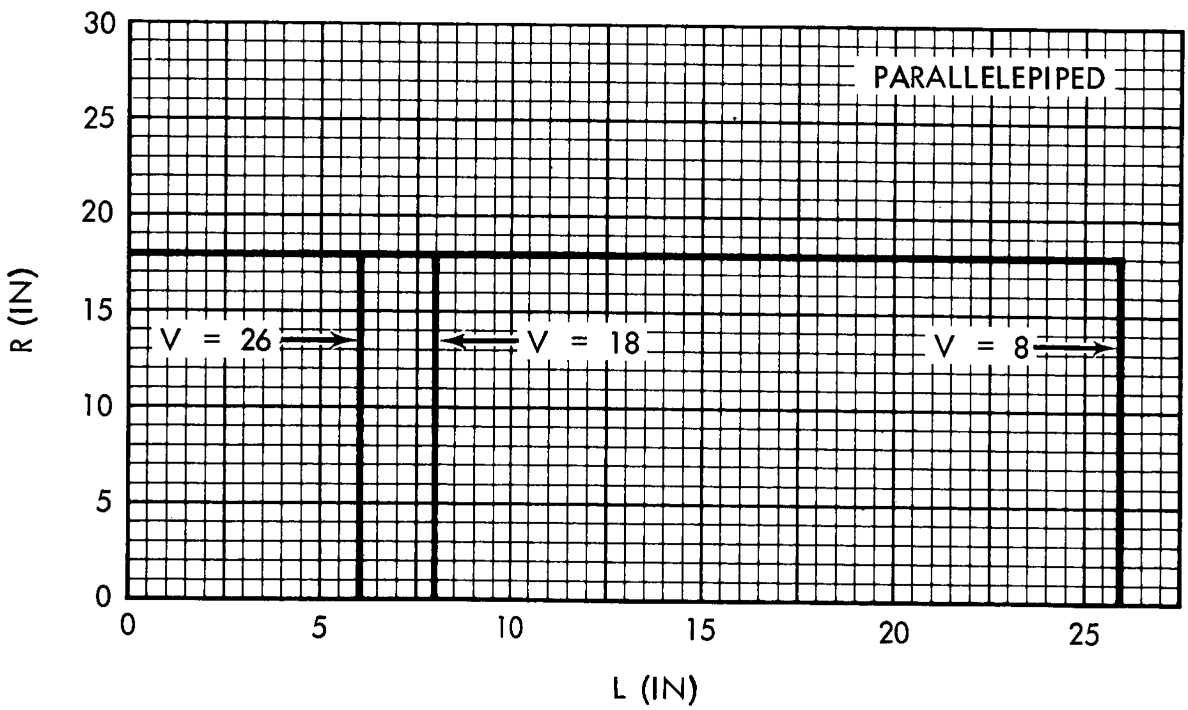
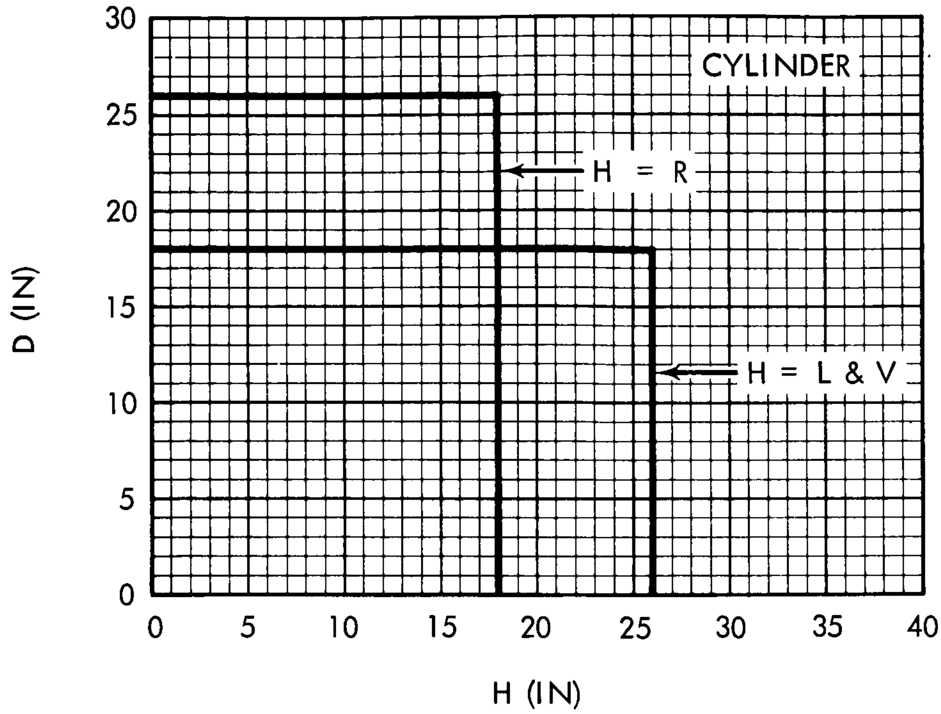


Figure A-31 STANDARD SHAPES CAPACITY - CAVITY 5-4



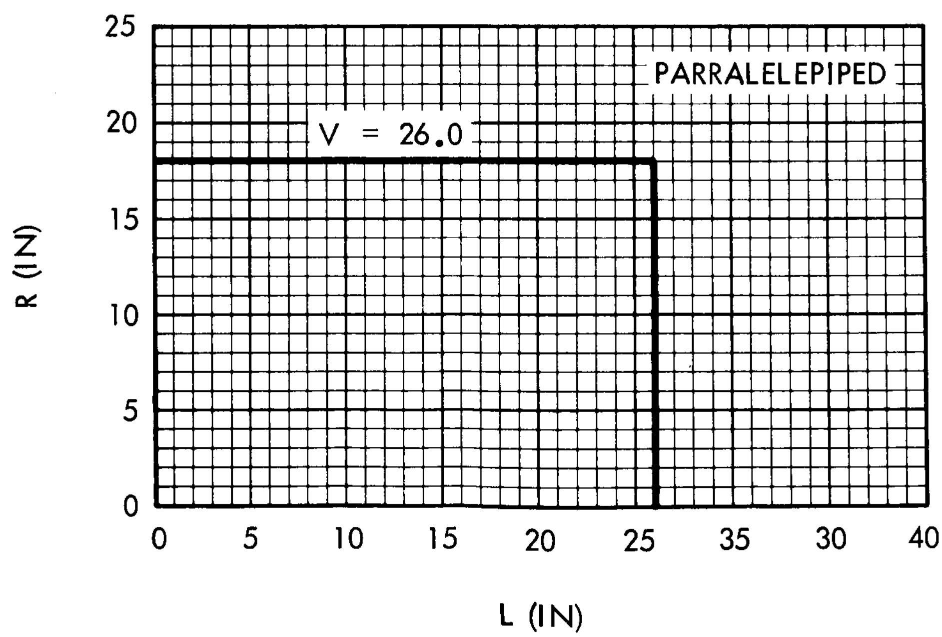
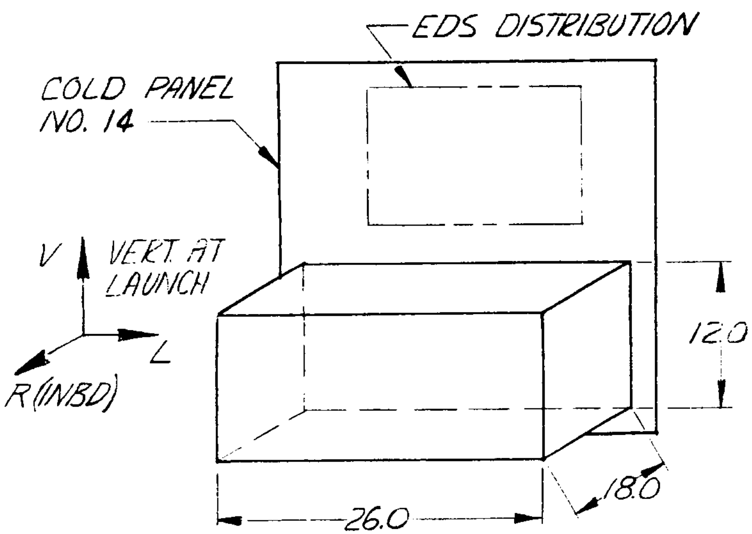


Figure A-32 STANDARD SHAPES CAPACITY - CAVITY 5-5

CAVITY 5-6

LOCATION: INSTRUMENT UNIT COLD PANEL NO. 14
EFFECTIVITY: SA-205 AND ON
VOLUME: 5,616 IN^3
WT. CAPACITY: 297 LBS.



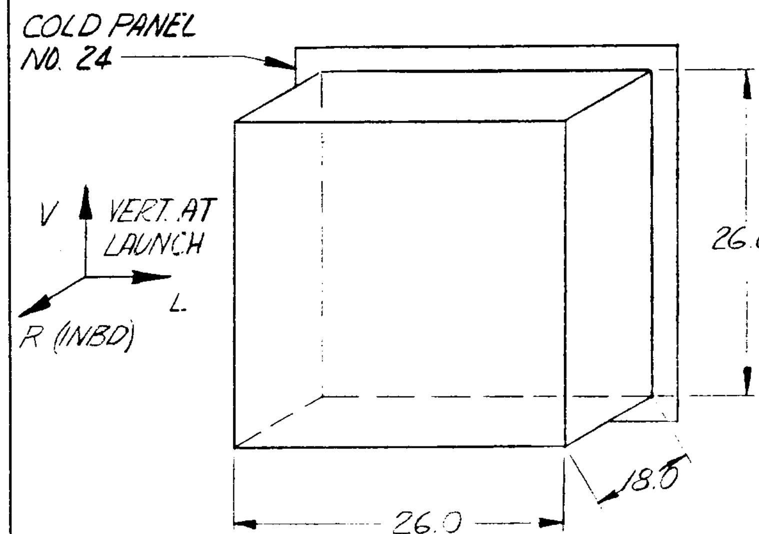
STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	12.0(V) x 18.0(R) x 26.0(L)
CYLINDER (H=V)	1	18.0(DIA) x 12.0(H)
CYLINDER (H=R)	1	12.0(DIA) x 18.0(H)
CYLINDER (H=L)	1	12.0(DIA) x 26.0(H)
SPHERE	1	12.0(DIA)

Figure A-33 DESCRIPTIVE DRAWING - CAVITY 5-6

CAVITY 5-7

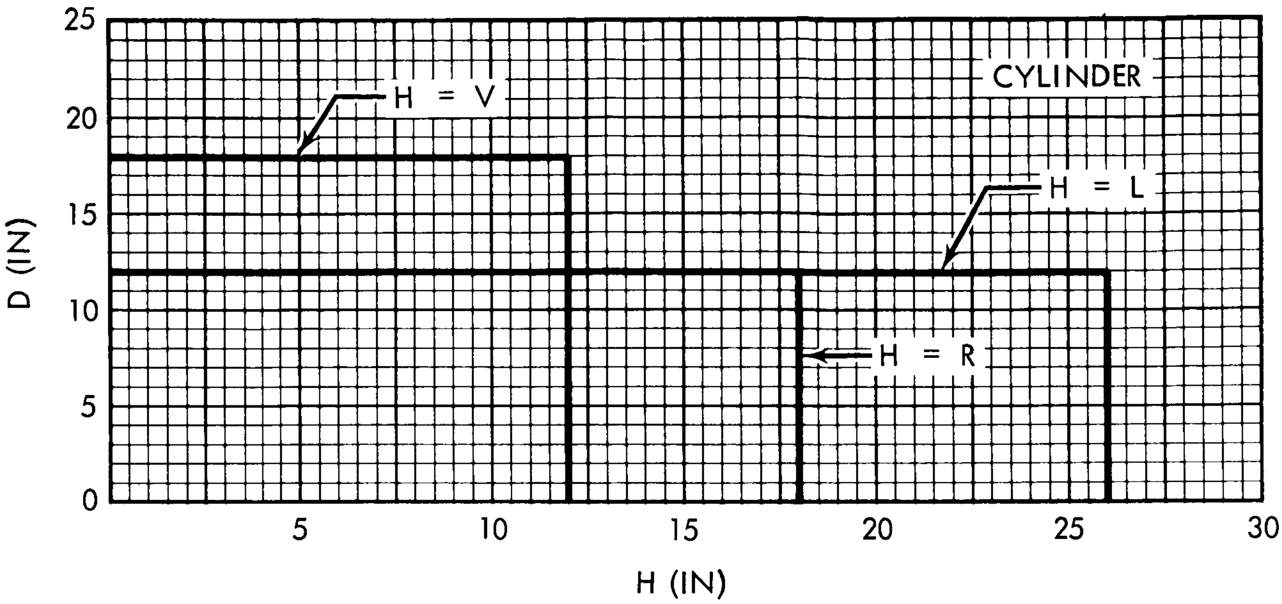
LOCATION: INSTRUMENT UNIT COLD PANEL NO. 24
EFFECTIVITY: SA-201, SA-202, SA-204 AND ON
VOLUME: 12,168 IN^3
WT. CAPACITY: 330 LBS.



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	26.0(V) x 18.0(R) x 26.0(L)
CYLINDER (H=V&L)	1	18.0(DIA) x 26.0(H)
CYLINDER (H=R)	1	26.0(DIA) x 18.0(H)
SPHERE	1	18.0(DIA)

Figure A-34 DESCRIPTIVE DRAWING - CAVITY 5-7



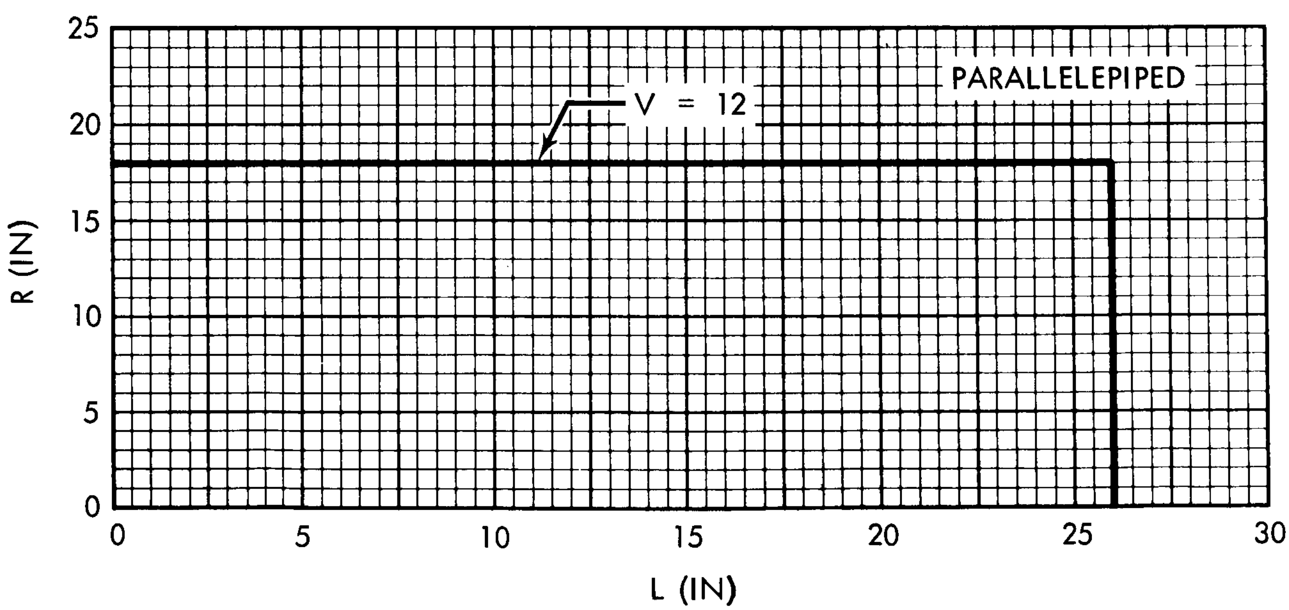
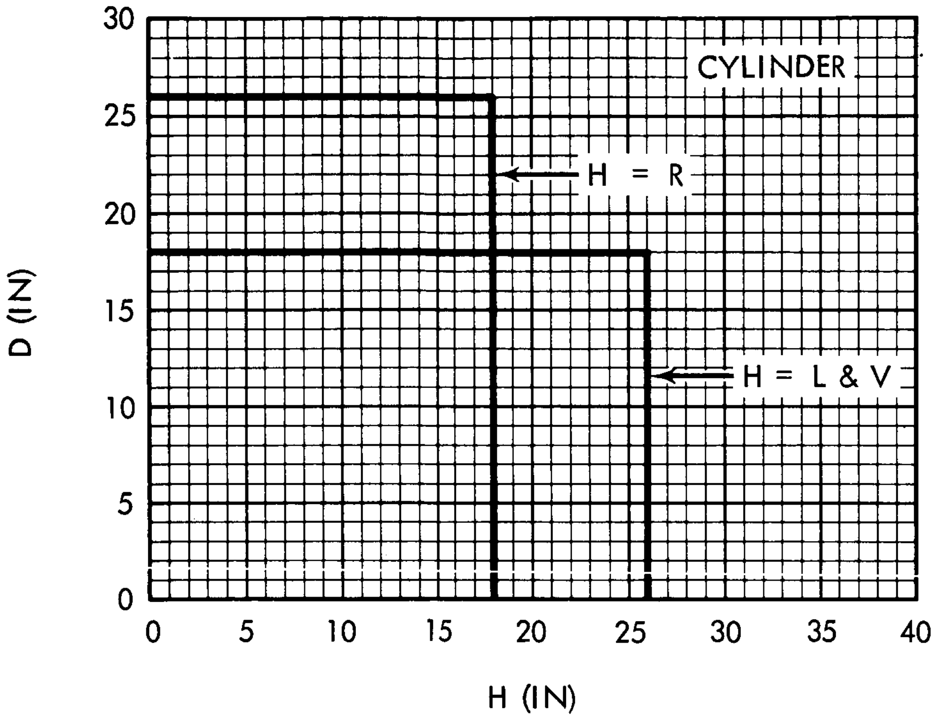


Figure A-35 STANDARD SHAPES CAPACITY - CAVITY 5-6



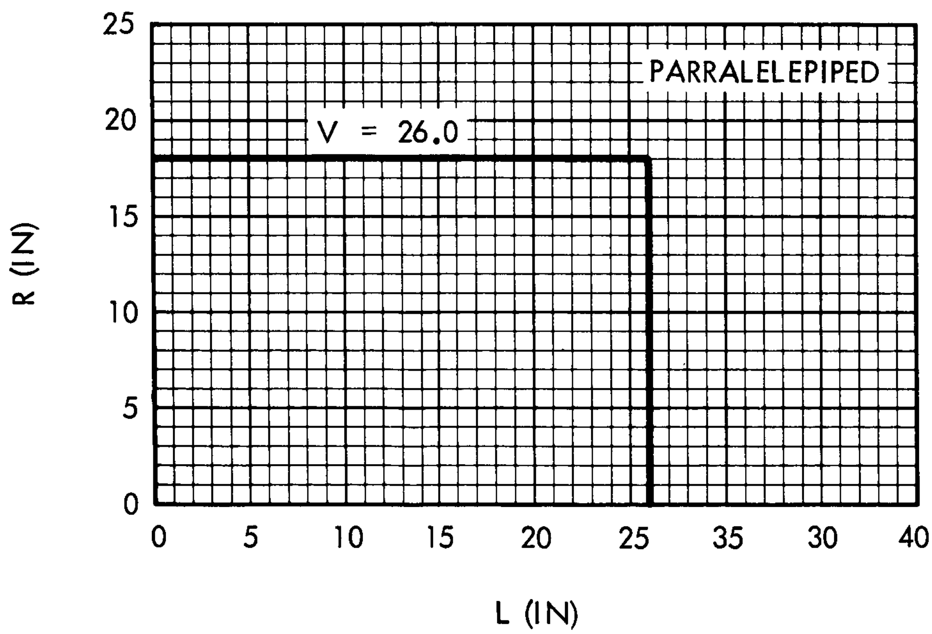
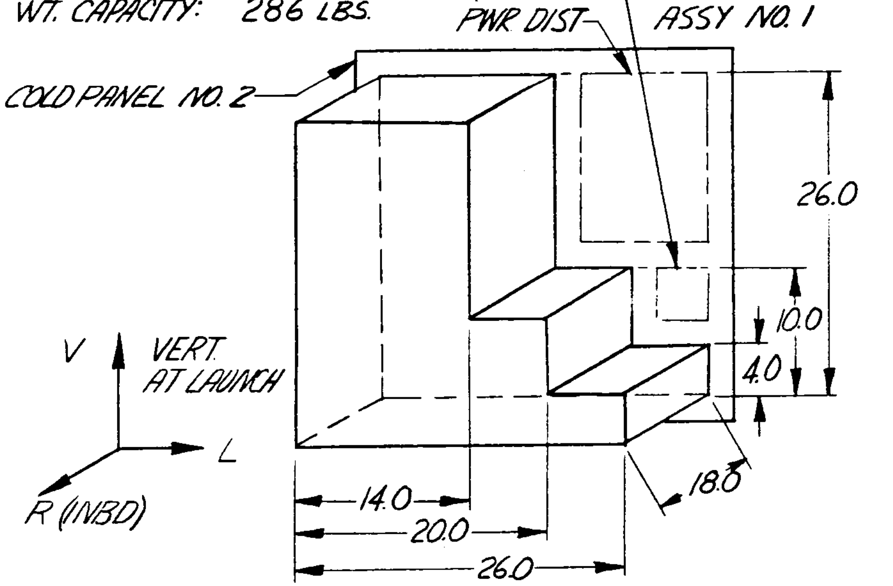


Figure A-36 STANDARD SHAPES CAPACITY - CAVITY 5-7

CAVITY 5-8

LOCATION: INSTRUMENT UNIT COLD PANEL NO. 2
EFFECTIVITY: SA-205 AND ON
VOLUME: 8,064 IN^3
WT. CAPACITY: 286 LBS.



STANDARD SHAPES CAPACITY

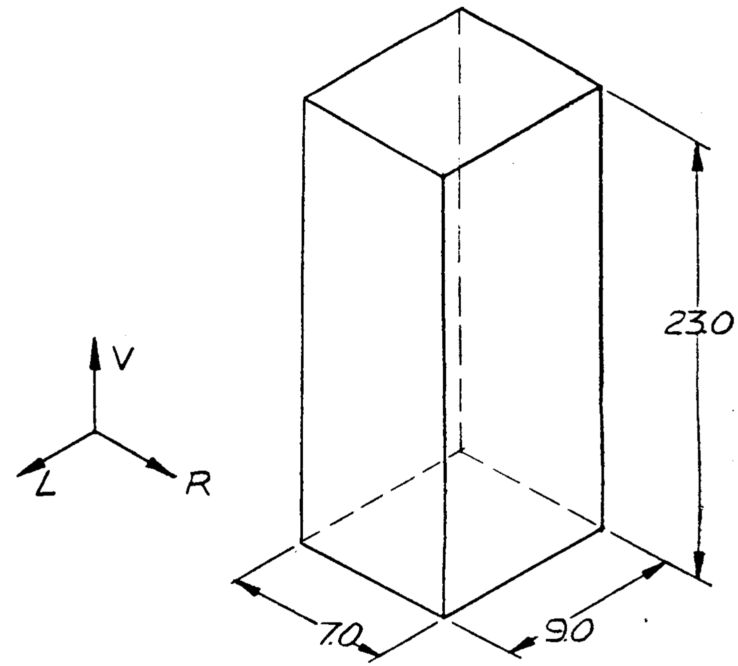
TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	26.0(V) × 18.0(R) × 14.0(L)
PARALLELEPIPED	1	10.0 (V) × 18.0(R) × 20.0(L)
PARALLELEPIPED	1	4.0 (V) × 18.0 (R) × 26.0(L)
CYLINDER H=V	1	14.0(DIA) × 26.0 (H)
CYLINDER H=V	1	18.0(DIA) × 10.0 (H)
CYLINDER H=L	4	4.0(DIA) × 26.0 (H)
CYLINDER H=R	1	14.5(DIA) × 18.0(H)
CYLINDER H=L	1	18.0(DIA) × 14.0(H)
CYLINDER H=L	1	10.0(DIA) × 20.0(H)
SPHERE	1	14.5 (DIA)

BROOKS 9-10-65

Figure A-37 DESCRIPTIVE DRAWING - CAVITY 5-8

CAVITY 6-1

LOCATION: S-IVB FWD SKIRT COLD PANEL NO. 12
EFFECTIVITY: SA-201
VOLUME: 1,449 IN^3
MASS CAPACITY: 50LBS



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	23.0(V) × 7.0(R) × 9.0(L)
CYLINDER (H=V)	1	7.0(DIA) × 23.0(H)
CYLINDER (H=R)	2	9.0 (DIA) × 7.0 (H)
CYLINDER (H=L)	3	7.0 (DIA) × 9.0 (H)
SPHERE	3	7.0 (DIA)

Figure A-38 DESCRIPTIVE DRAWING - CAVITY 6-1

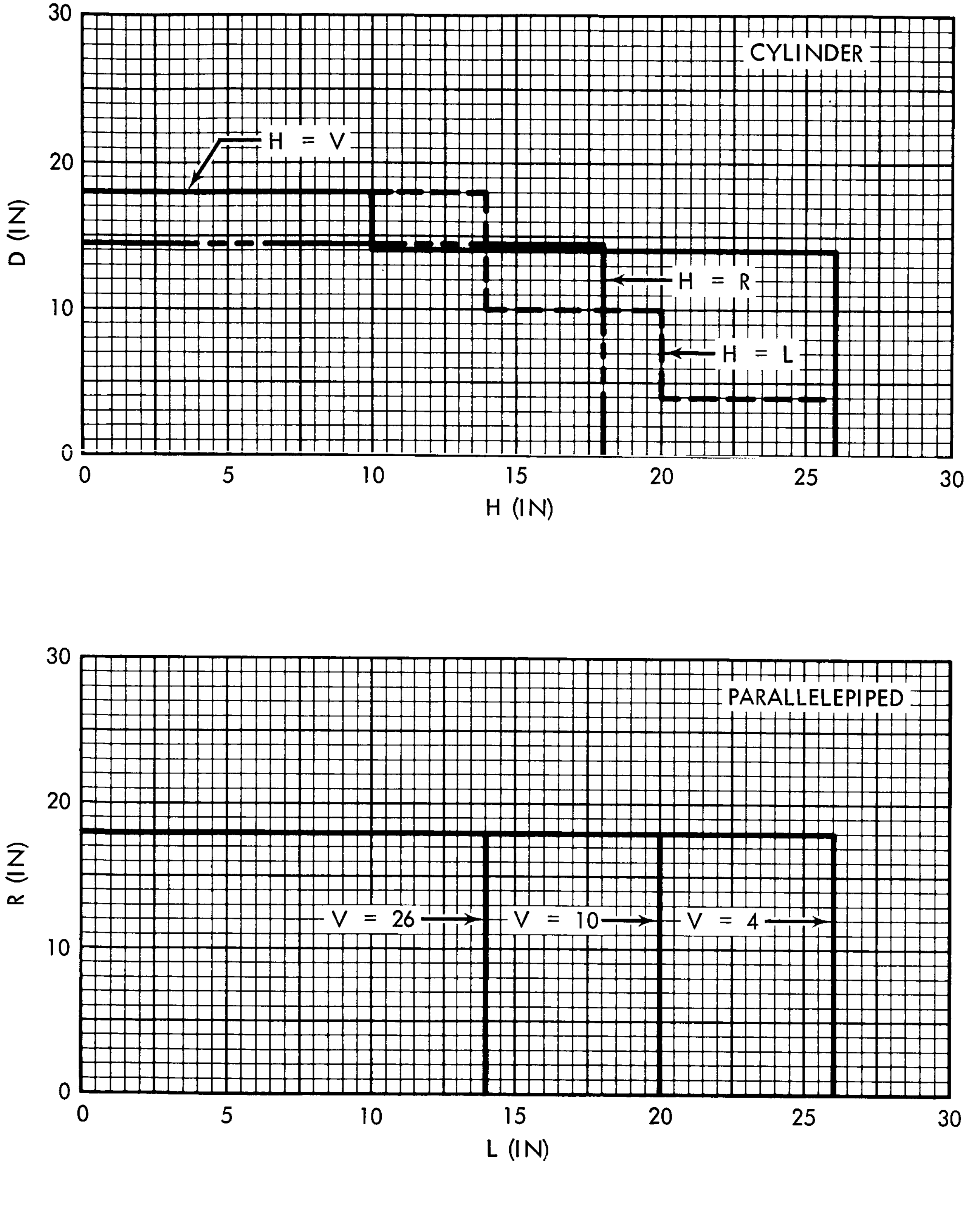
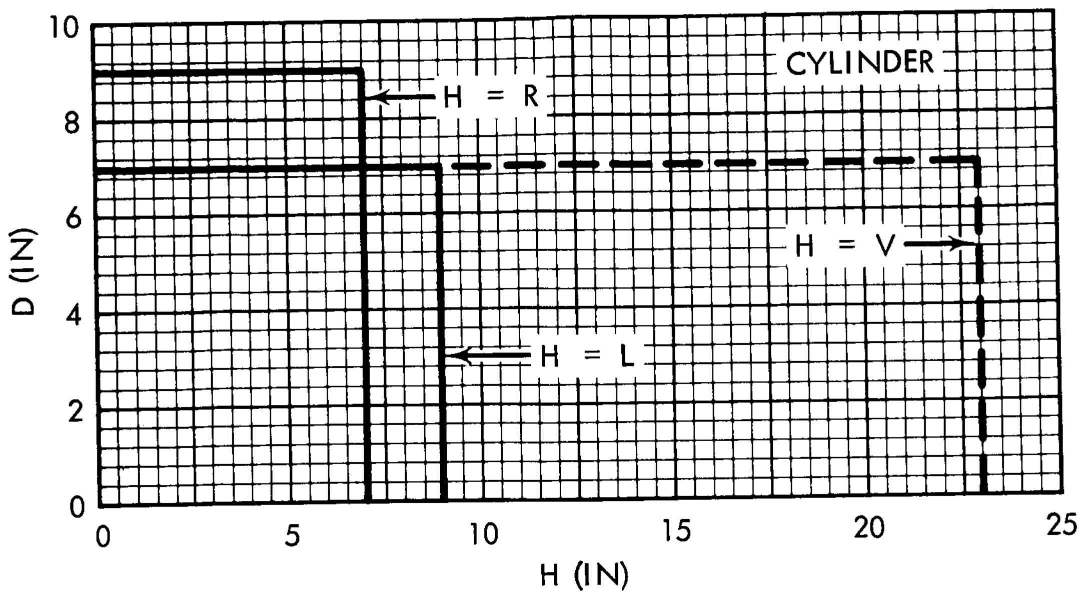


Figure A-39 STANDARD SHAPES CAPACITY - CAVITY 5-8



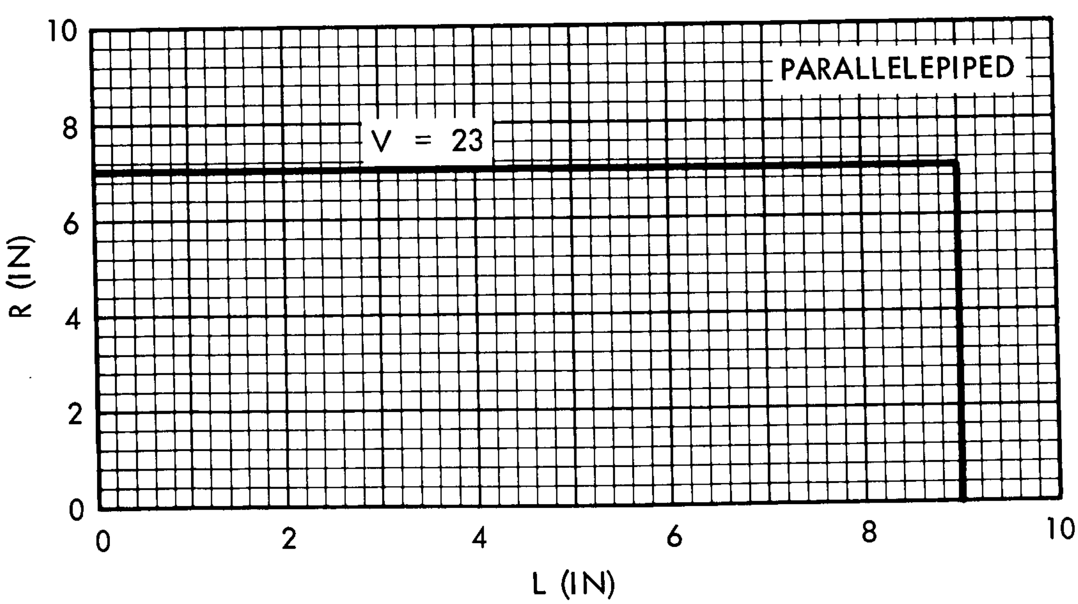
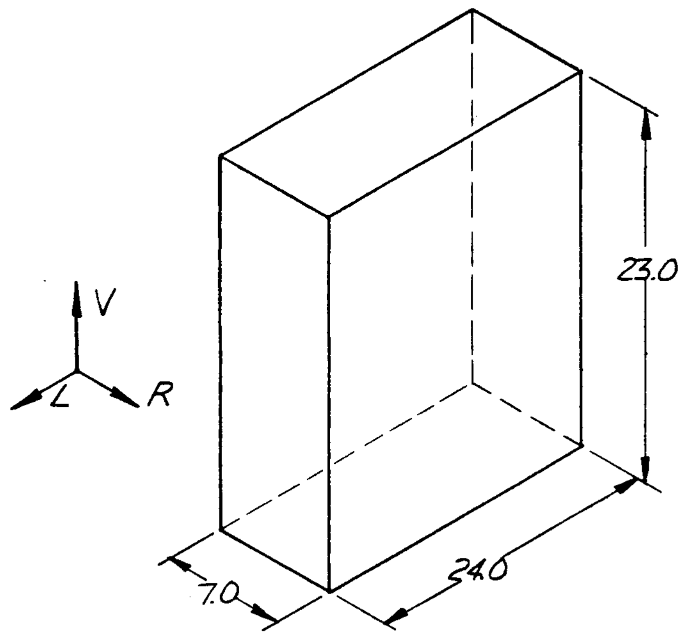


Figure A-40 STANDARD SHAPES CAPACITY - CAVITY 6-1

CAVITY 6-2

LOCATION: S-IVB FWD SKIRT COLD PANEL NO. 7
EFFECTIVITY: SA-201
VOLUME: 3,864 IN^3
MASS CAPACITY: 130 LBS



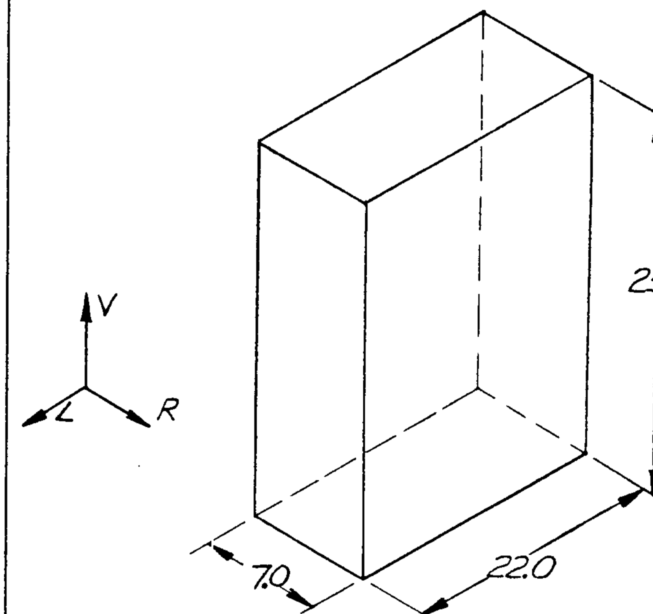
STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	23.0(V)× 7.0(R)× 24.0(L)
CYLINDER (H=V)	3	7.0(DIA)× 23.0(H)
CYLINDER (H=R)	1	23.0(DIA)× 7.0(H)
CYLINDER (H=L)	3	7.0(DIA)× 24.0(H)
SPHERE	9	7.0(DIA)

Figure A-41 DESCRIPTIVE DRAWING - CAPACITY 6-2

CAVITIES 6-3 & 6-5

LOCATION: S-IVB FWD SKIRT COLD PANEL NOS. 6 & 4
EFFECTIVITY: SA-201
VOLUME: 3,542 IN^3
MASS CAPACITY: 125 LBS



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	23.0(V)× 7.0(R)× 22.0(L)
CYLINDER (H=V)	3	7.0(DIA)× 23.0(H)
CYLINDER (H=R)	1	22.0(DIA)× 7.0(H)
CYLINDER (H=L)	3	7.0(DIA)× 22.0(H)
SPHERE	9	7.0(DIA)

BROOKS 11-26-65

Figure A-42 DESCRIPTIVE DRAWING - CAVITIES 6-3 & 6-5

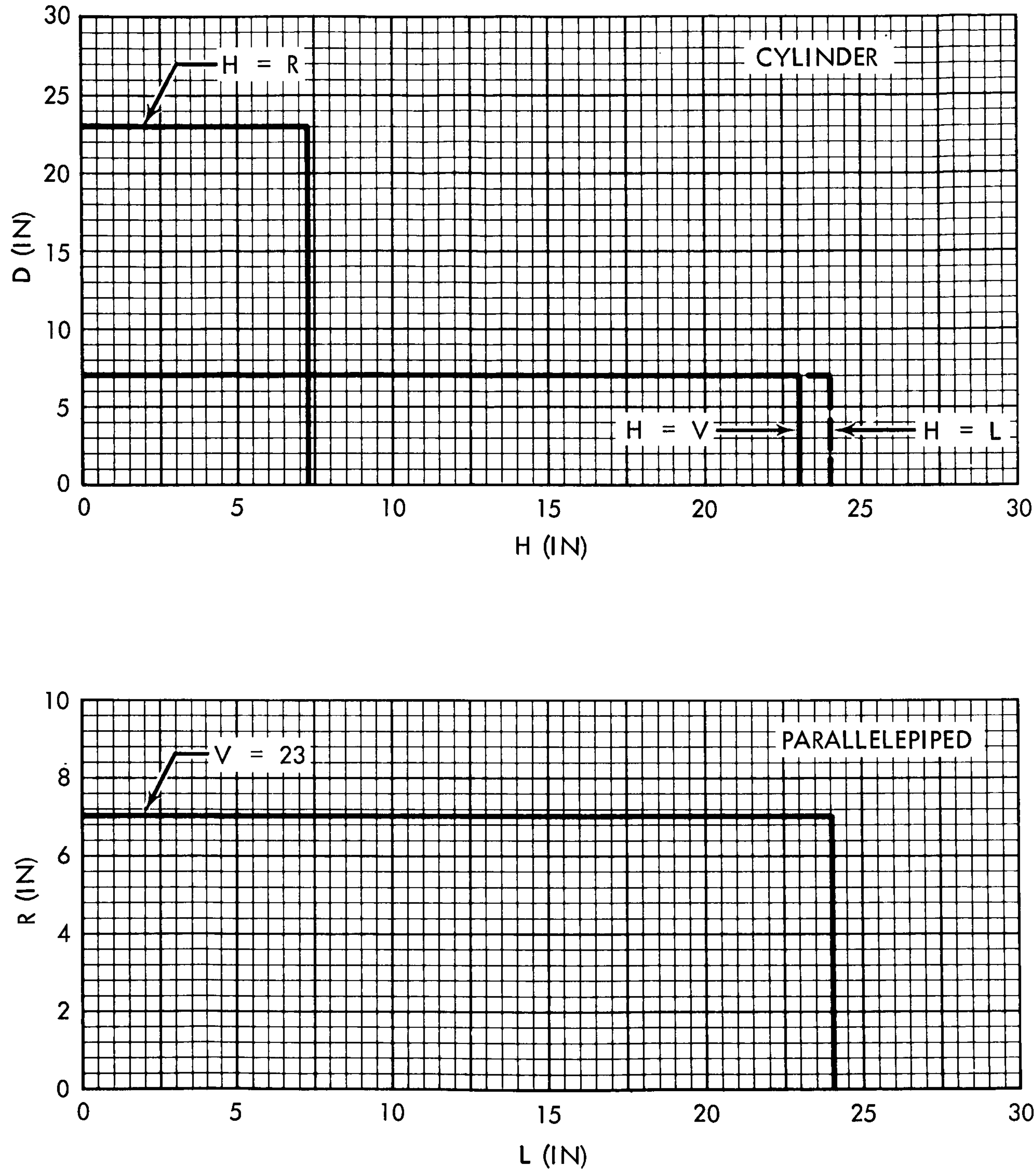
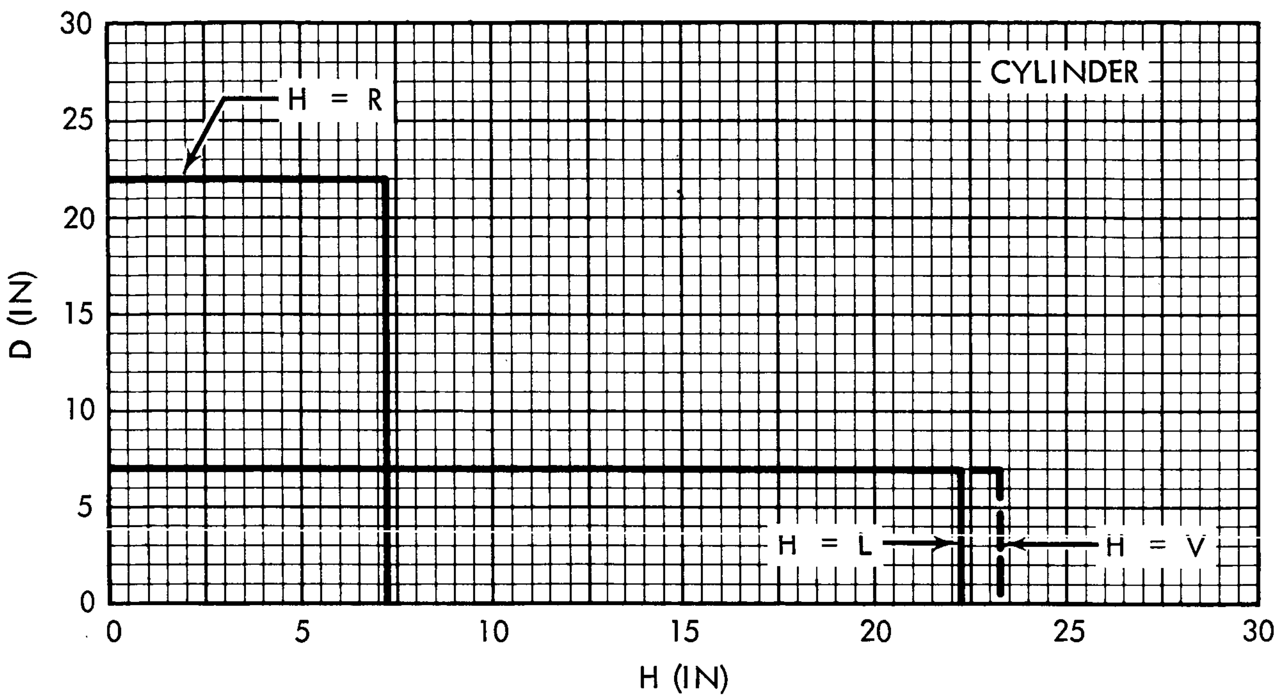


Figure A-43 STANDARD SHAPES CAPACITY - CAVITY 6-2



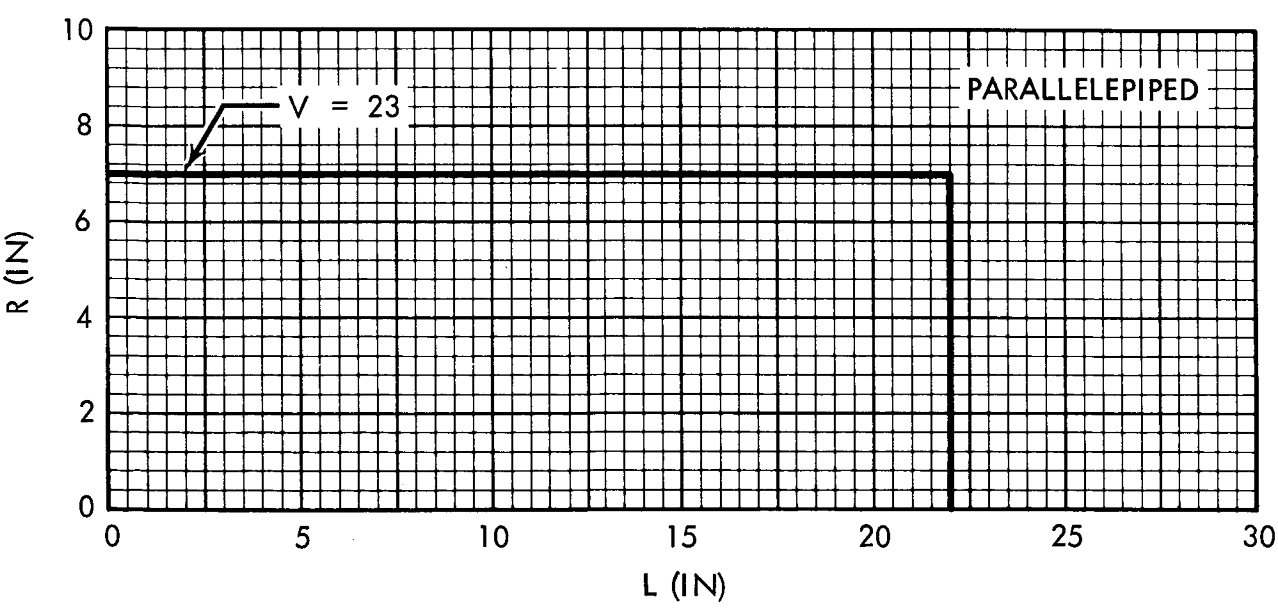
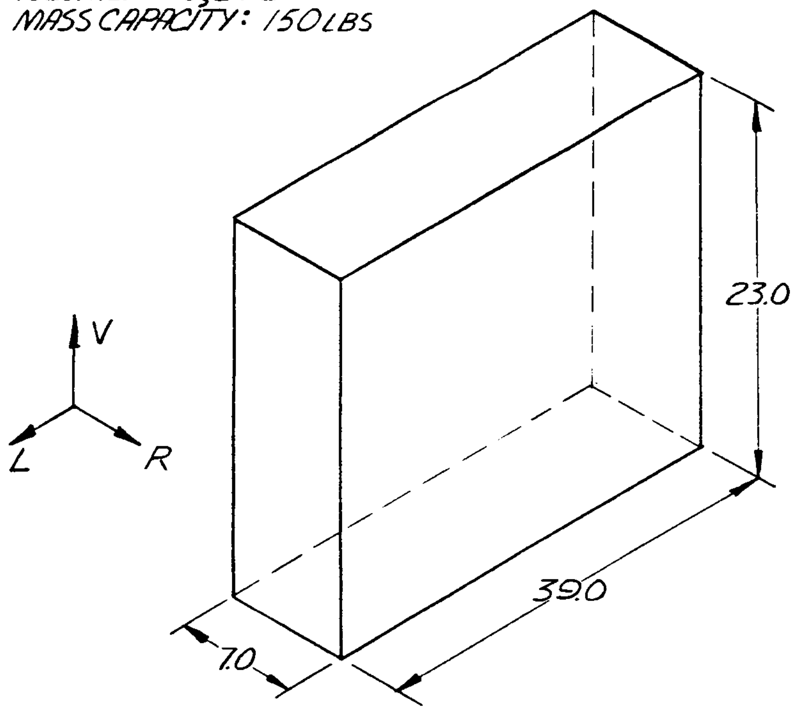


Figure A-44 STANDARD SHAPES CAPACITY - CAVITIES 6-3 & 6-5

CAVITY 6-4

LOCATION: S-IVB FWD SKIRT COLD PANEL NO. 5
EFFECTIVITY: SA-201
VOLUME: 6,279 IN^3
MASS CAPACITY: 150 LBS



STANDARD SHAPES CAPACITY

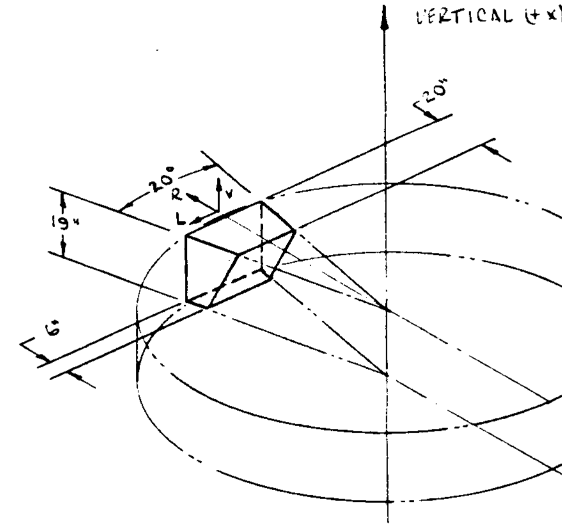
TYPE	NO.	SIZE (INCHES)
PARALLELEPIPED	1	23.0(V) × 7.0(R) × 39.0(L)
CYLINDER (H=V)	5	7.0(DIA) × 23.0(H)
CYLINDER (H=R)	1	23.0(DIA) × 7.0(H)
CYLINDER (H=L)	3	7.0(DIA) × 39.0(H)
SPHERE	15	7.0(DIA)

BROOKS 11-30-65

Figure A-45 DESCRIPTIVE DRAWING - CAVITY 6-4

CAVITIES 7-1 THRU 7-18

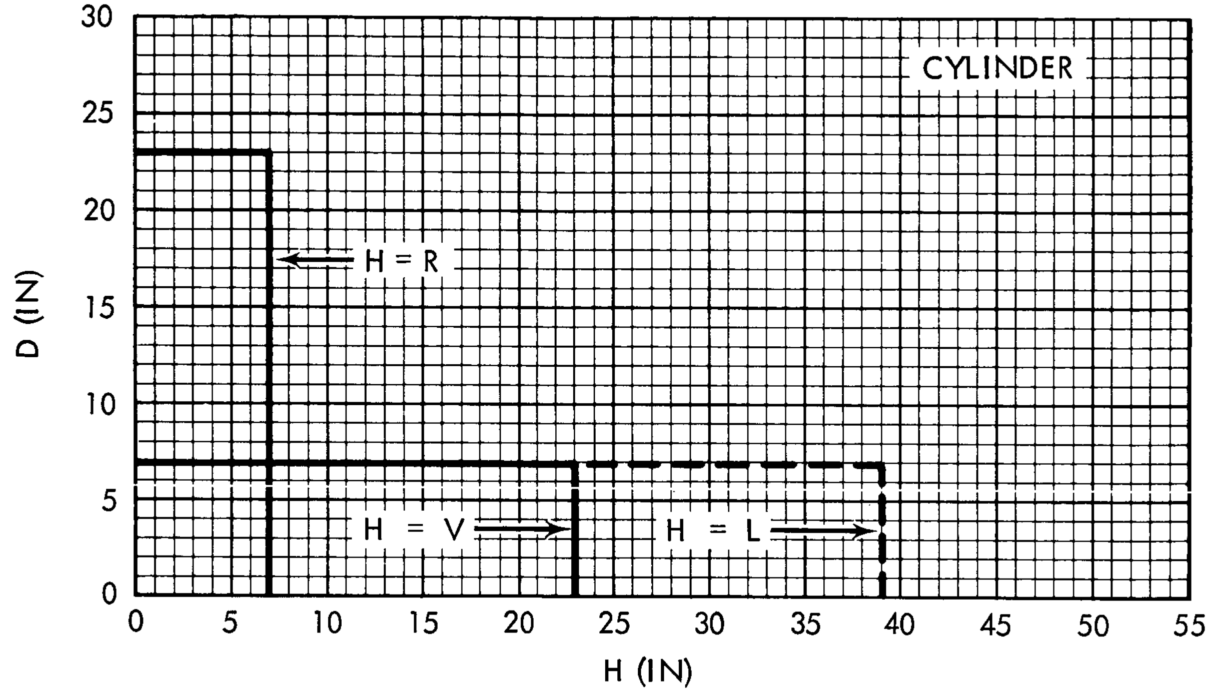
LOCATION: BELOW WORK PLATFORM (SIVB/IU REGION)
EFF: SA-207 AND ON
VOL: 9610 IN^3 each
WT. CAP: 1,000 LBS



STANDARD SHAPES CAPACITY

TYPE	NO.	SIZE ~ INCHES
PARALLELEP.		SEE ATTACHED SHEET
CYLINDER		SEE ATTACHED SHEET
SPHERE	1	13.2 DIA

Figure A-46 DESCRIPTIVE DRAWING - CAVITIES 7-1 THRU 7-18



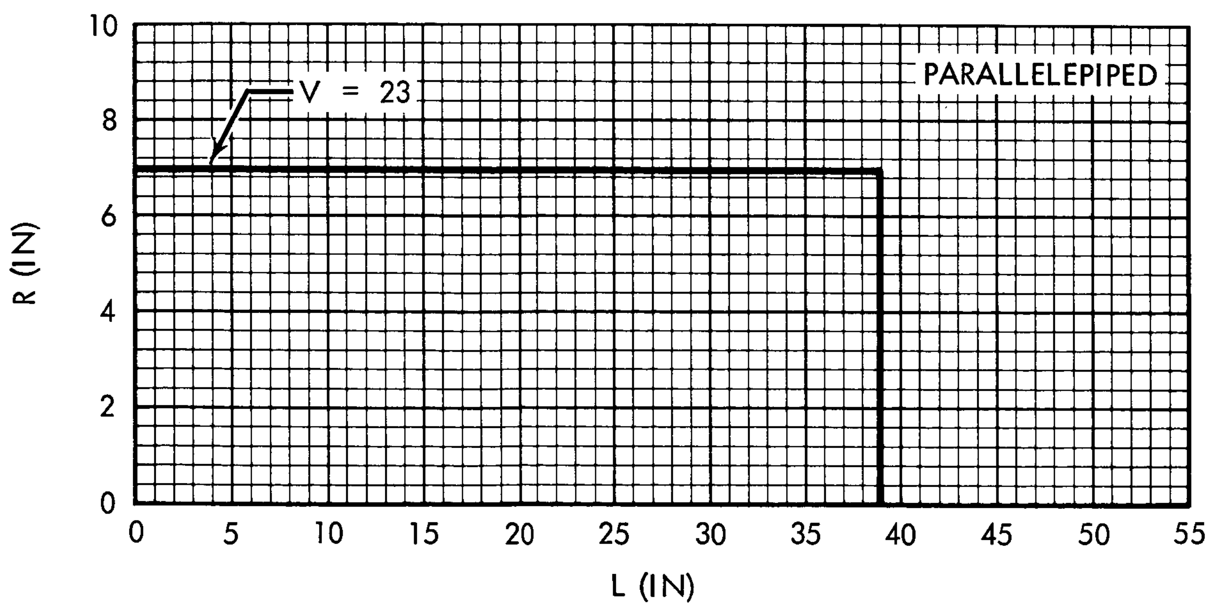
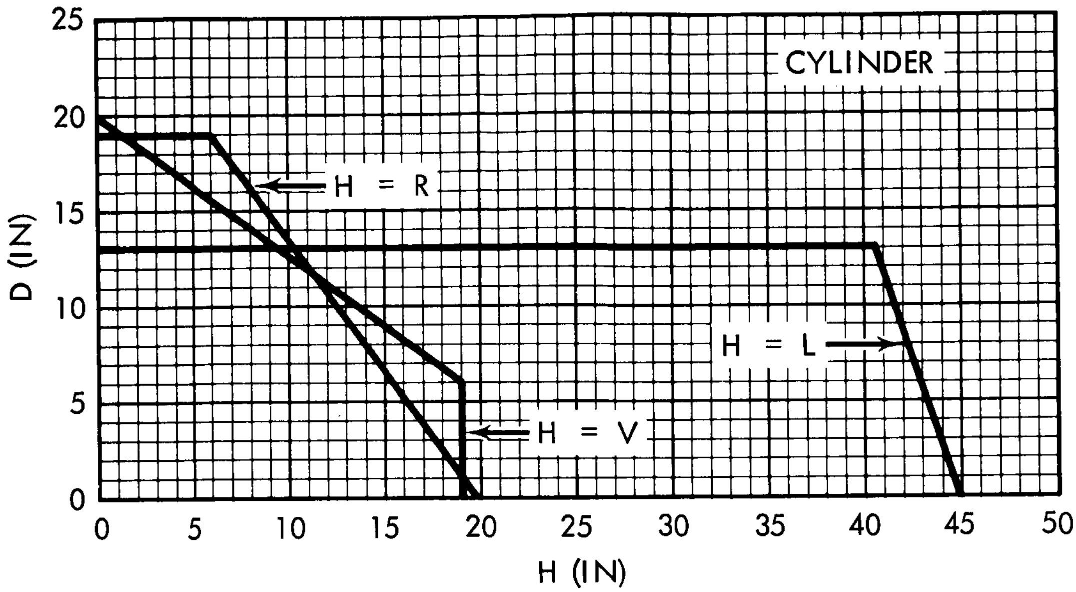


Figure A-47 STANDARD SHAPES CAPACITY - CAVITY 6-4



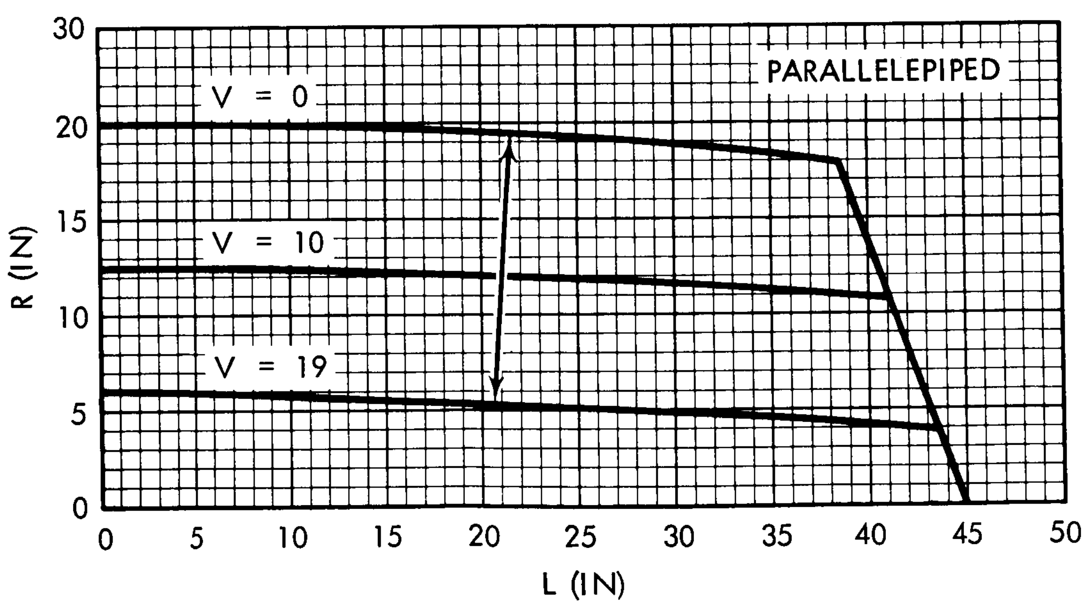


Figure A-48 STANDARD SHAPES CAPACITY - CAVITIES 7-1 THRU 7-18

APPENDIX B

EXPERIMENT EFFECTIVENESS DATA

This section contains orbital lifetime data and effectiveness definitions generated during the experiment/mission effectiveness analyses discussed in Section 5.

B.1 ORBITAL LIFETIME DATA

Basic orbital lifetime data, generated for the maximum operational capability configuration of the SIVB/IU vehicle (i.e., Apollo payload separated, fairings deployed), are shown in Figure B-1. In computing the ballistic coefficient of 58.6 kg/m^2 (12 lb/ft^2) for this configuration, a broadside spin of the vehicle was assumed. The assumption of broadside spin yields conservative estimates of orbital lifetime. General Dynamics computer Procedure F26, "Satellite Vehicle Performance Program," was used to compute lifetimes of short duration (less than a few days) by numerical integration of the equations of motion. Lifetimes of longer duration were estimated using the analytical approximations of King Hele (Reference 5.1). The 1959 ARDC Atmosphere was used in all orbital perturbation analyses. Orbital lifetimes of deployed experiments were approximated by multiplying the lifetime data of Figure B-1 by the ratio of ballistic coefficients.

Ballistic coefficients of the deployed experiments (Deployment Modes 3 and 4) were approximated for the representative experiment configurations described in Volume II. A summary of ballistic coefficient approximations for the representative experiment configurations is shown in Table B-1. The approximations were based on either maximum, minimum, or average cross-sectional area, depending upon the expected flight orientation. Also included in Table B-1 is a ballistic parameter, $M/V^{2/3}$, which was found to correlate with the ballistic coefficient. This correlation led to the development of the following ballistic coefficient prediction formulae:

$$\beta = 173.9 \frac{M}{V^{2/3}} - 17.39 \text{ (attitude controlled)}$$

$$\beta = 86.95 \frac{M}{V^{2/3}} - 8.695 \text{ (random tumbling)}$$

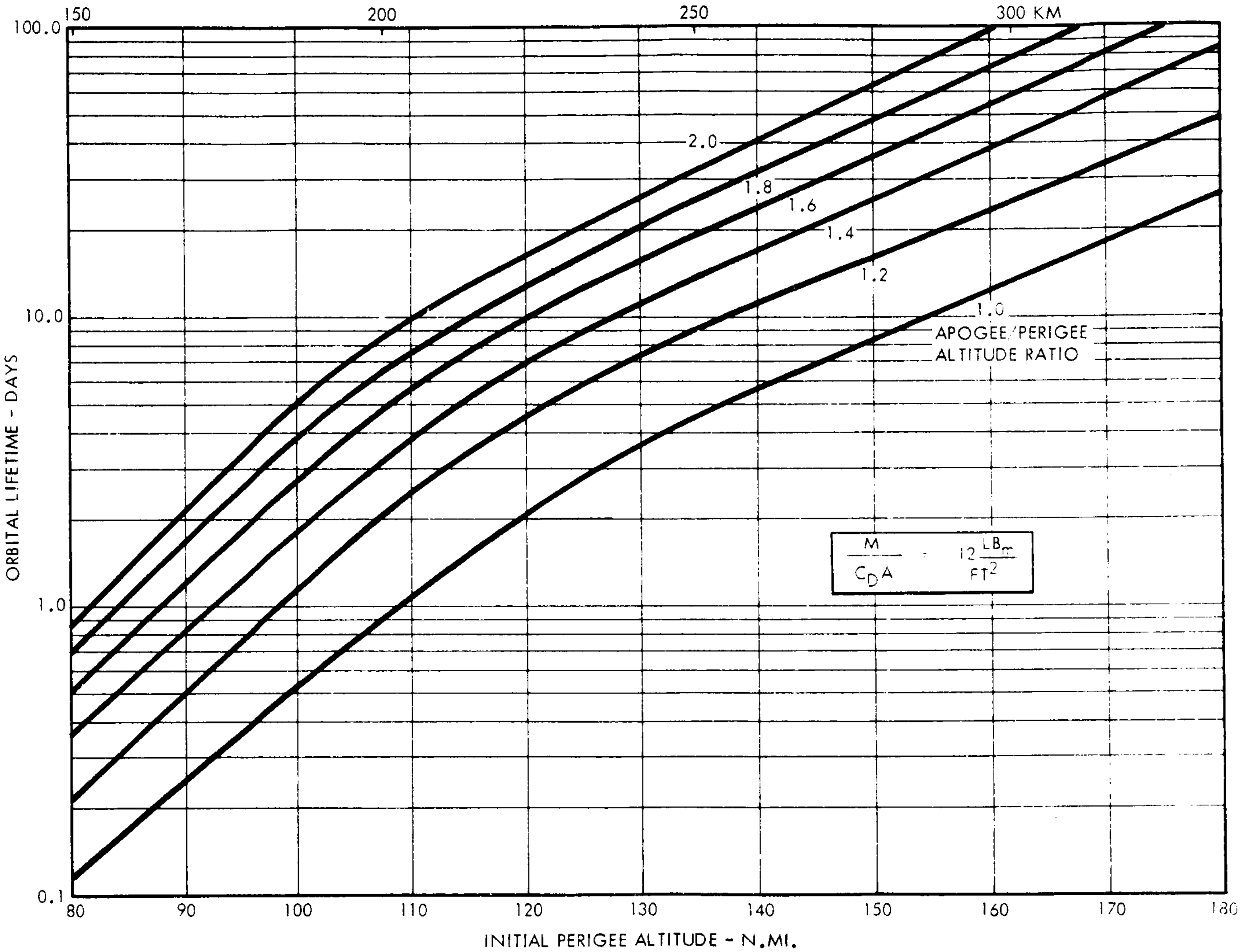


Figure B-1 ORBITAL LIFETIMES

TABLE B-1
SUMMARY OF REPRESENTATIVE BALLISTIC COEFFICIENTS

CATEGORY	NO.	VOLUME in^3 (Total)	D	W (lb.)	$\frac{W}{C_D A_{Max}}$	$\frac{W}{C_D A_{Min}}$	$\left(\frac{W}{C_D A}\right)_{Ave}$	$\frac{W}{V^{2/3}}$
MI	1	81,200	53.7	983.6		88.71^A		.53
MI	2	39,468	42.3	814.6		93.27^A		.70
MI	3	139,000	64.3	1310.1		71.46^A		.49
MI	4	79,083	53.3	730.0		61.11^A		.40
MI	5	469,800	96.5	2561.5		62.10^A		.42
SLG	3	102,993	58.2	648.1	9.1	13.2	11.2^T	.30
M	1	17,400	32.2	212.6	7.5	38.0	22.8^T	.32
M	3	8,700	25.6	186.1	18.2	35.7	27.0^T	.44
M	4	13,696	29.7	123.0	7.8	16.1	11.9^T	.22
M	5	8,757	25.6	207.0	18.6	39.6	29.1^T	.49
SDT	1	122,000	61.6	937.1	8.8			.38
SDT	2	3,689	19.2	53.0	1.4	1.4	1.4^T	.23
SDT	3	72,900	51.9	635.5		57.1^A		.37
SDT	4	47,595	45.0	443.0		36.1^A		.34
SDT	5	30,300	38.7					
MS	1	73,500	52.1	722.8	19.1	45.2	32.1^T	.41
OEA	2	19,600	33.5	284.0		40.7^A		.39
OEA	4	22,140	34.9	364.5		45.4^A		.46
OEA	3	17,512	32.3	344.0		61.1^A		.51
OEA	5	31,870	39.4	631.8		103.3^A		.63

A - Attitude controlled experiments; use minimum frontal area.

T - Tumbing experiments;use average ballistic coefficient.

where

β = ballistic coefficient

M = mass of experiment after deployment

V = volume of experiment after deployment.

These formulae can be used to obtain "first-pass" estimates of the ballistic coefficients of amorphous experiments. Predicted and computed ballistic coefficients for 23 experiments are compared in Figure B-2.

B.2 EXPERIMENT EFFECTIVENESS DEFINITIONS

Effectiveness definitions for 20 selected experiments (defined in Volume II) are presented on the following pages. The data for each experiment consist of three components: (1) basic effectiveness definitions, (2) final effectiveness definitions, and (3) completed effectiveness library work sheets.

The basic effectiveness definitions were formulated as functions of the parameters which influence experiment effectiveness. They were, for the most part, defined subjectively by the experimenter and/or mission analyst after analysis of the data acquisition objectives of the experiment. Final effectiveness definitions were obtained by relating basic experiment effectiveness to one (or two) of the initial orbital elements and mission parameters listed in Table 5-2.

In some cases, the basic effectiveness of an experiment was defined directly by the experimenter as a function of the initial orbital elements and/or mission parameters and is, therefore, a final effectiveness definition. The final effectiveness data for each of the 20 experiments were entered into the Effectiveness Library Work Sheets as described in Section 3.2 of Volume IV. Utilization of the work sheets is a convenient intermediate step for the transfer of final effectiveness definitions into the Experimental Payload Characteristics Library of SEPTER.

Illustrations of the effectiveness definitions and applicable library work sheets are presented as Figures B-3 through B-73 in this appendix. A cross reference of this material is provided in Table B-2.

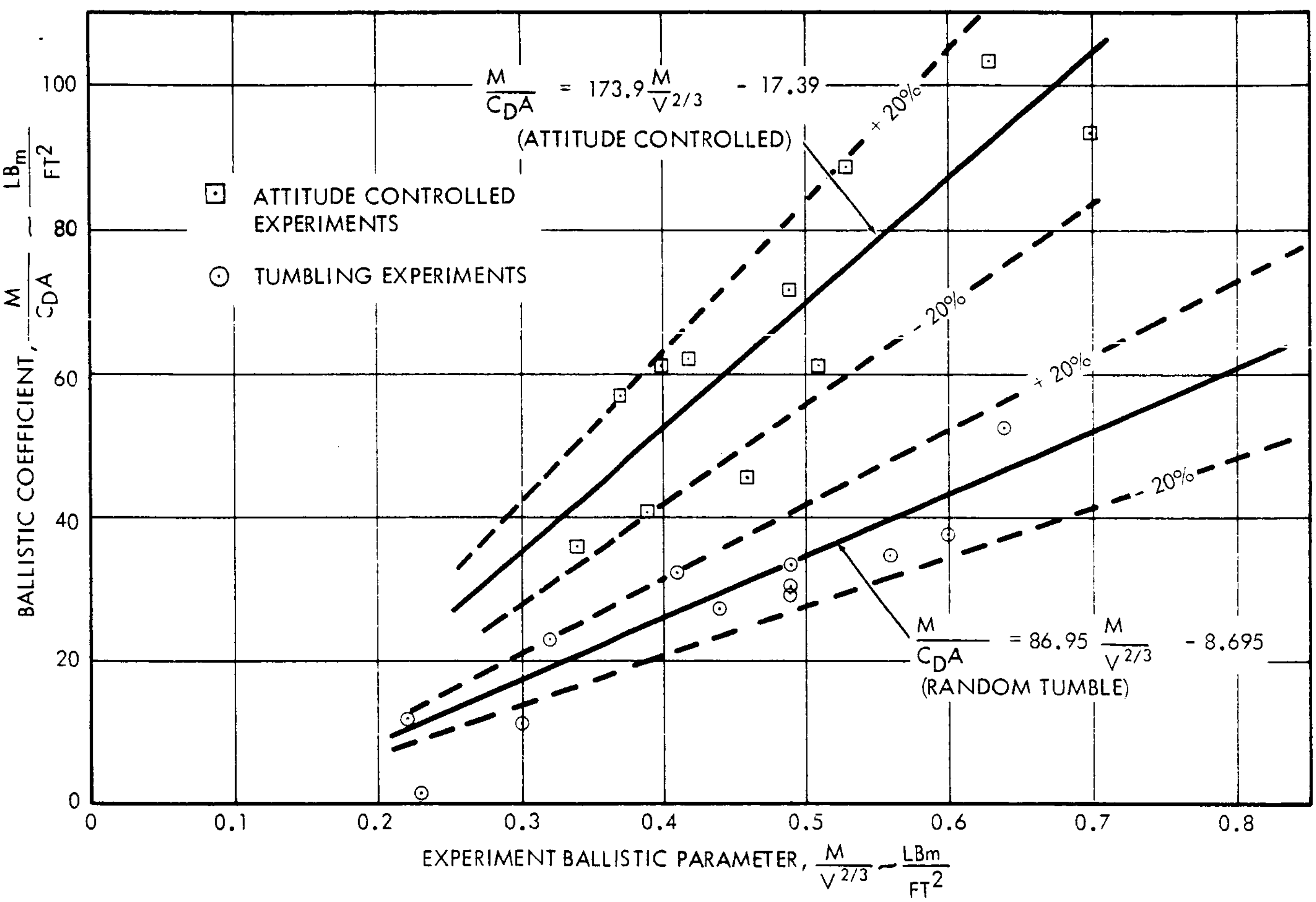


Figure B-2 BALLISTIC COEFFICIENT PREDICTION FOR AMORPHOUS EXPERIMENTS

TABLE B-2
SUMMARY OF EXPERIMENT EFFECTIVENESS DEFINITIONS AND WORK SHEETS

EXPERIMENT		FIGURE NUMBERS		
CATEGORY	NO.	BASIC DEFINITIONS	FINAL DEFINITIONS	EXPERIMENT LIBRARY WORK SHEETS
SDT	4	B-3, B-4, B-5	B-6, B-7	B-8
SDT	5		B-9	B-10
MS	3	B-11, B-12, B-13	B-14	B-15
MS	4	B-16	B-17	B-18
MI	1	B-19	B-20,B-21,B-22, B-23,B-24,B-25	B-26
MI	2	B-19	B-22,B-23,B-24, B-25	B-27
SLG	1	B-28	B-29	B-30
SLG	2	B-31	B-32	B-33
SLG	4	B-28	B-34	B-35
SLG	5	B-36	B-37	B-38
M	1	B-39	B-40	B-41
M	2	B-42	B-43	B-44
M	3	B-45	B-46	B-47
M	4	B-48	B-49	B-50
M	5	B-51	B-52	B-53
OEA	1	B-54	B-1,B-55,B-56,B-57	B-58
OEA	2	B-59	B-60,B-61,B-62	B-63
OEA	3		B-64,B-65	B-66
OEA	4	B-67	B-68,B-69,B-70	B-71
OEA	5	B-19	B-72	B-73

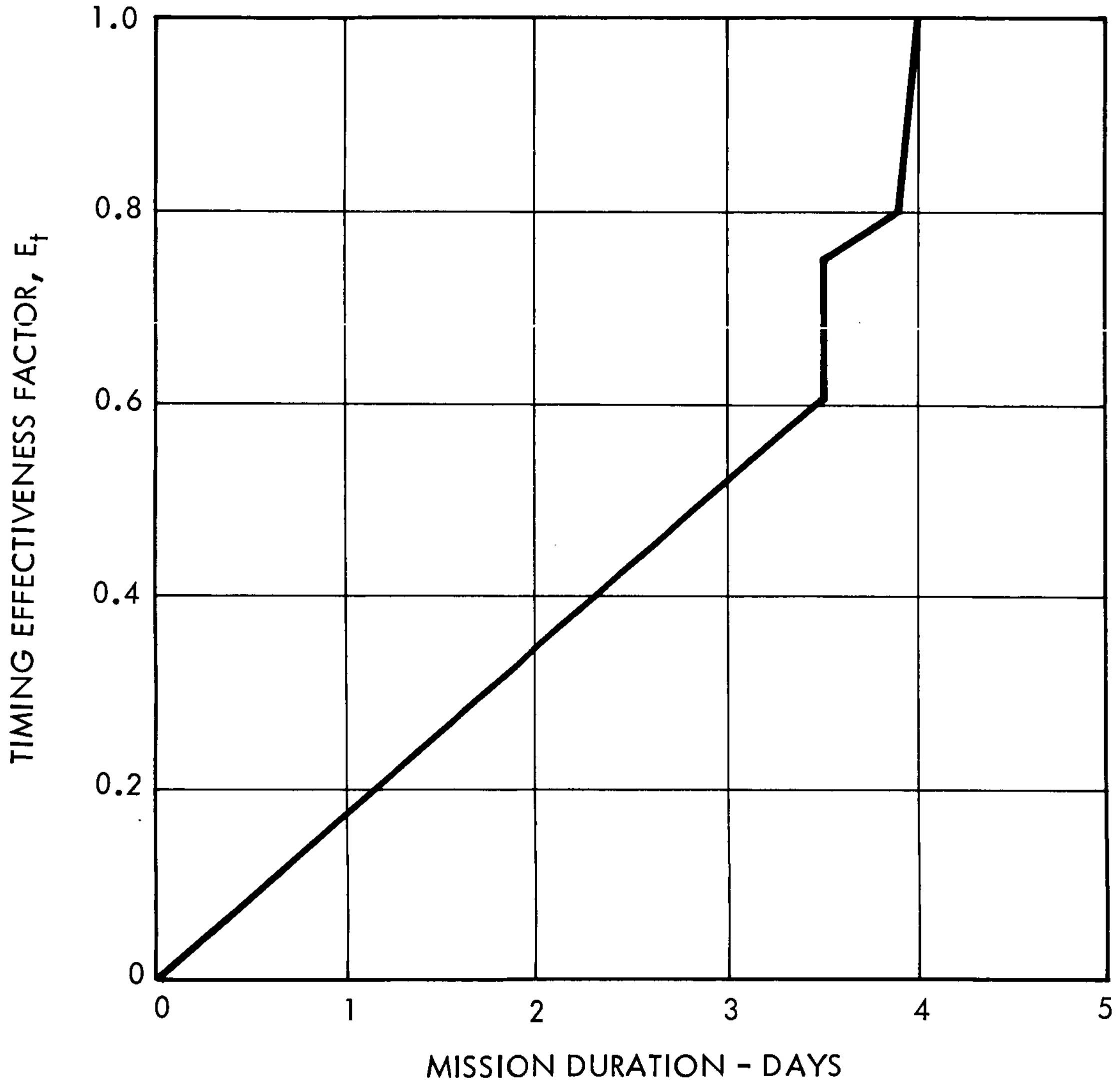


Figure B-3 BASIC EFFECTIVENESS DEFINITION -1, EXPERIMENT: SDT-4

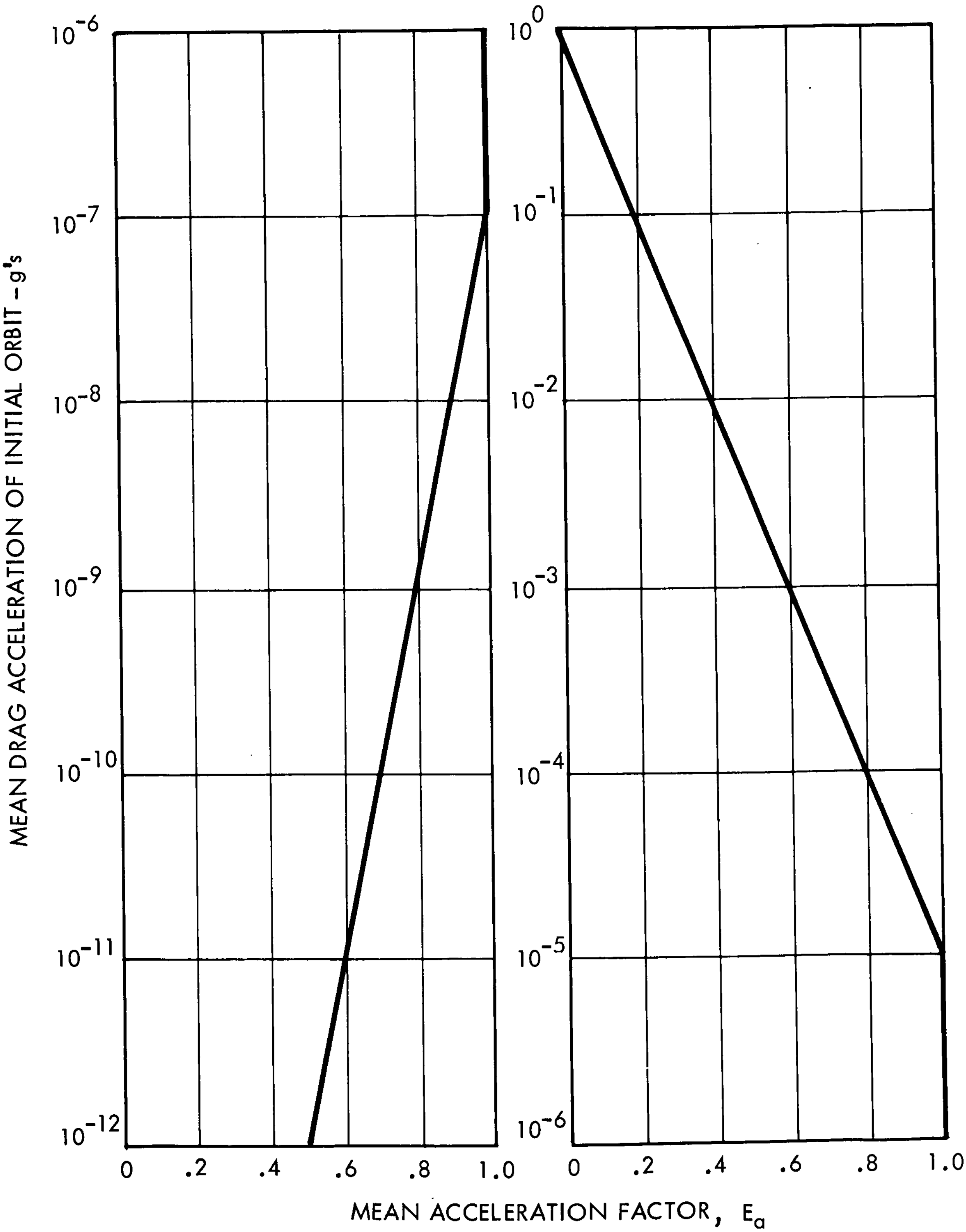


Figure B-4 BASIC EFFECTIVENESS DEFINITION - 2, EXPERIMENT: SDT-4

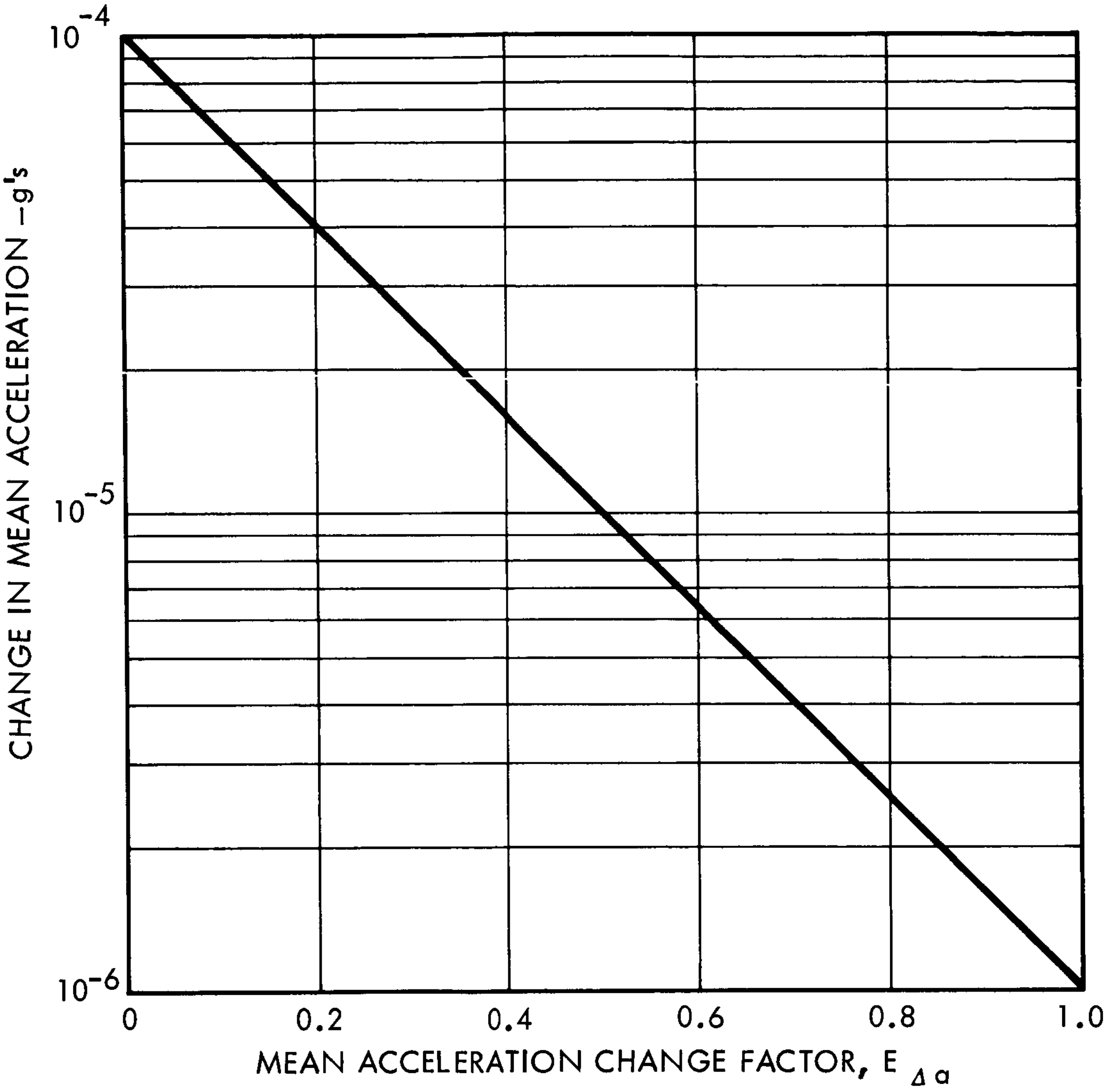


Figure B-5 BASIC EFFECTIVENESS DEFINITION - 3, EXPERIMENT: SDT-4

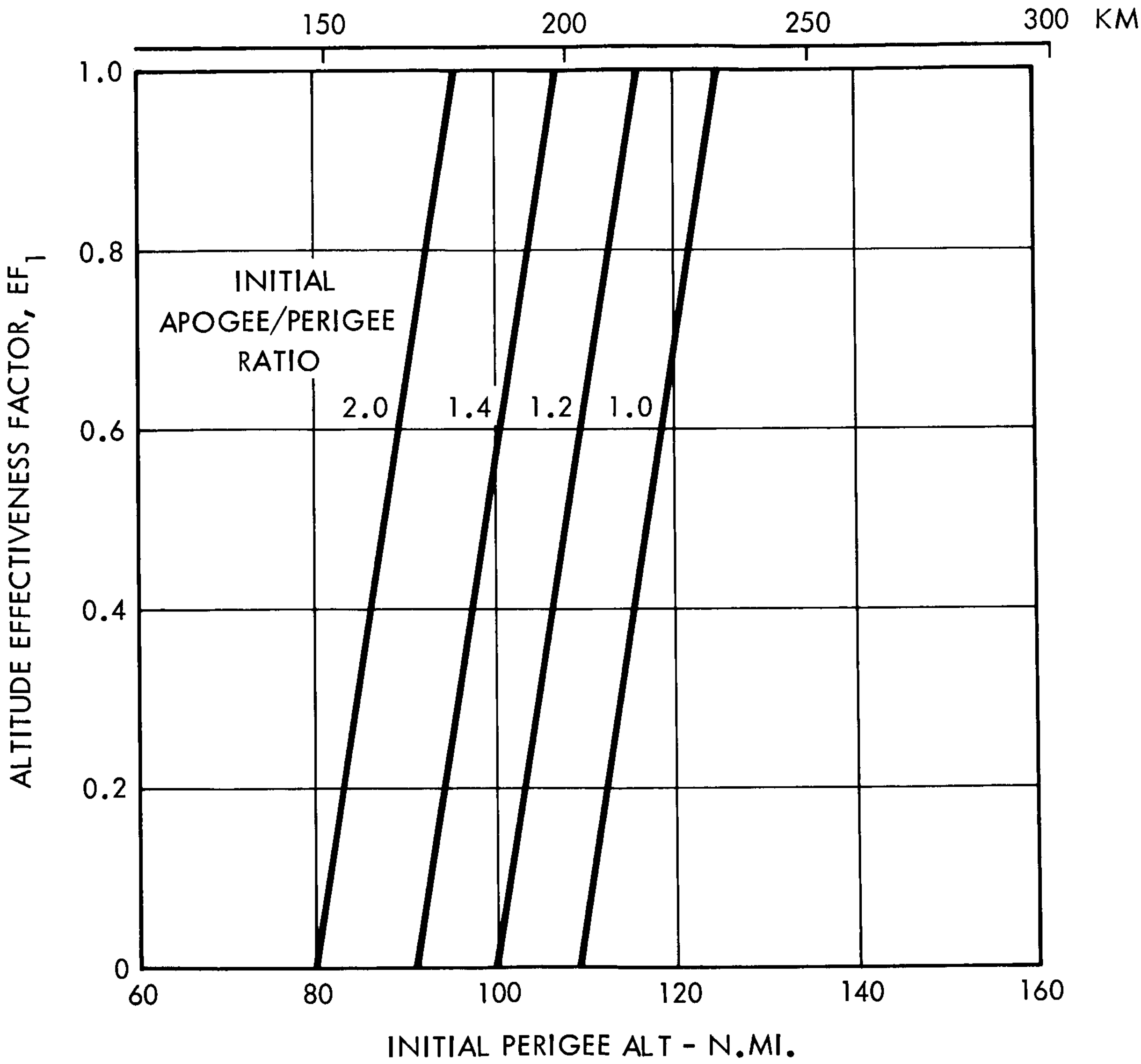


Figure B-6 FINAL EFFECTIVENESS DEFINITION - 1, EXPERIMENT: SDT-4

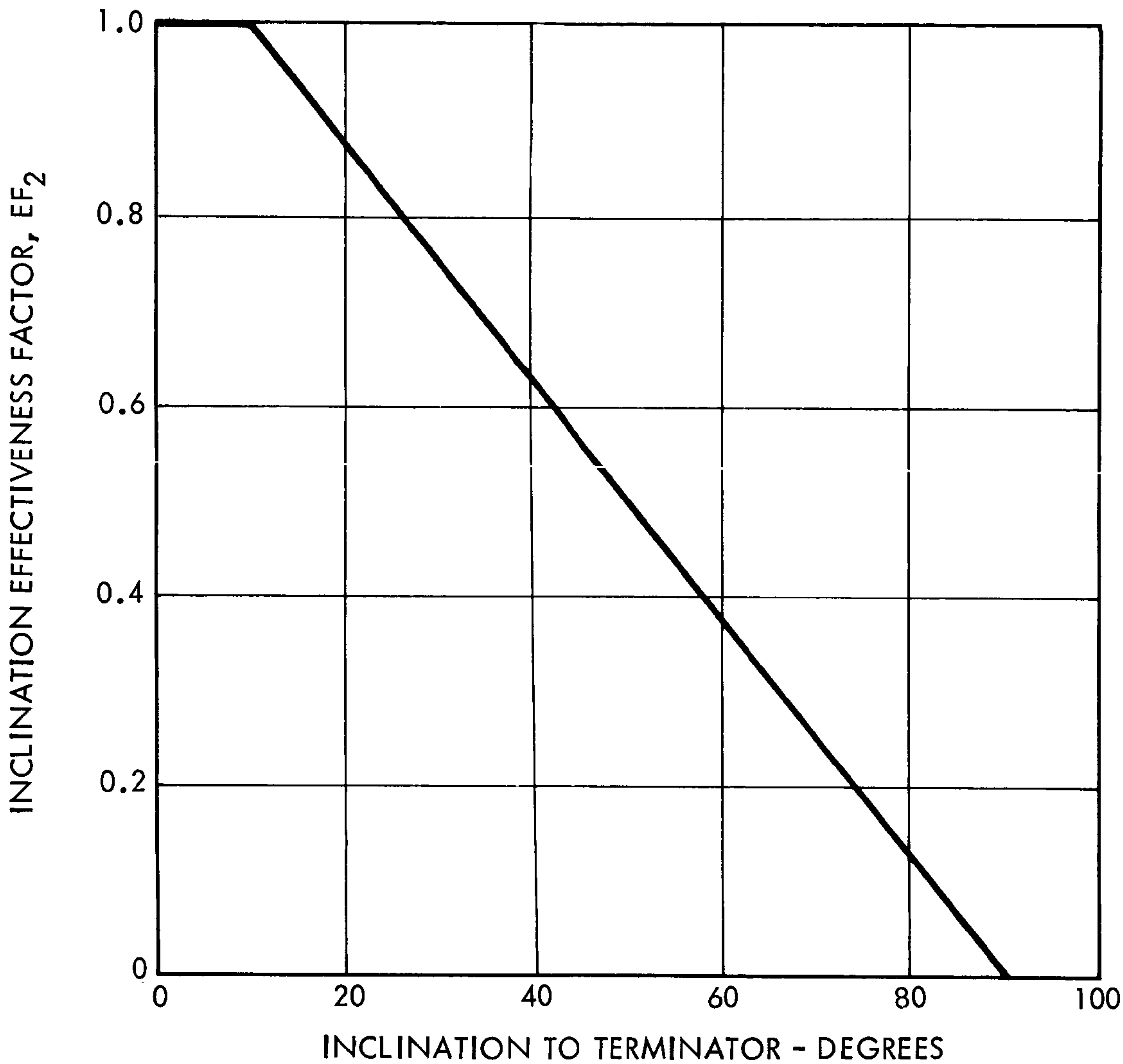


Figure B-7 FINAL EFFECTIVENESS DEFINITION - 2, EXPERIMENT: SDT-4

Figure B-8 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: SDT-4

EXPERIMENT: SDT-4 **IBM PROBLEM NO.** 1439P019

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	14	10								
Second Variable I.D.	(KY)	0	11								
Interp. Option	(KI)	1	1								
No. of Last Row	(IR)	2	10								
No. of First Column	(JC)	1	1								
No. of Abscissa Values	(NX)	5	8								
No. of Ordinate Values	(NY)	1	6								

EF2

INCLINATION TO TERMINATOR

EF1

APOGEE/PERIGEE ALT. RATIO

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	0.0	1.0	1.0	0.0	1.0	1.0									
2	0.0	0.0	10.0	90.0	170.0	180.0									
3															
4	2.0	-0.63	0	0.63	1.26	2.52	3.78	5.04	6.30						
5	1.8	-0.70	-0.07	0.56	1.19	2.45	3.71	4.97	6.23						
6	1.6	-0.95	-0.32	0.31	0.94	2.20	3.46	4.72	5.98						
7	1.4	-1.33	-0.70	-0.07	0.56	1.82	3.08	4.34	5.60						
8	1.2	-1.89	-1.26	-0.63	0	1.26	2.52	3.78	5.04						
9	1.0	-1.84	-1.21	-0.58	0.05	0.68	1.31	2.57	3.83						
10	0.0	129.64	148.16	166.68	185.2	222.24	259.28	296.32	333.36						
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

PERIGEE ALT. - km

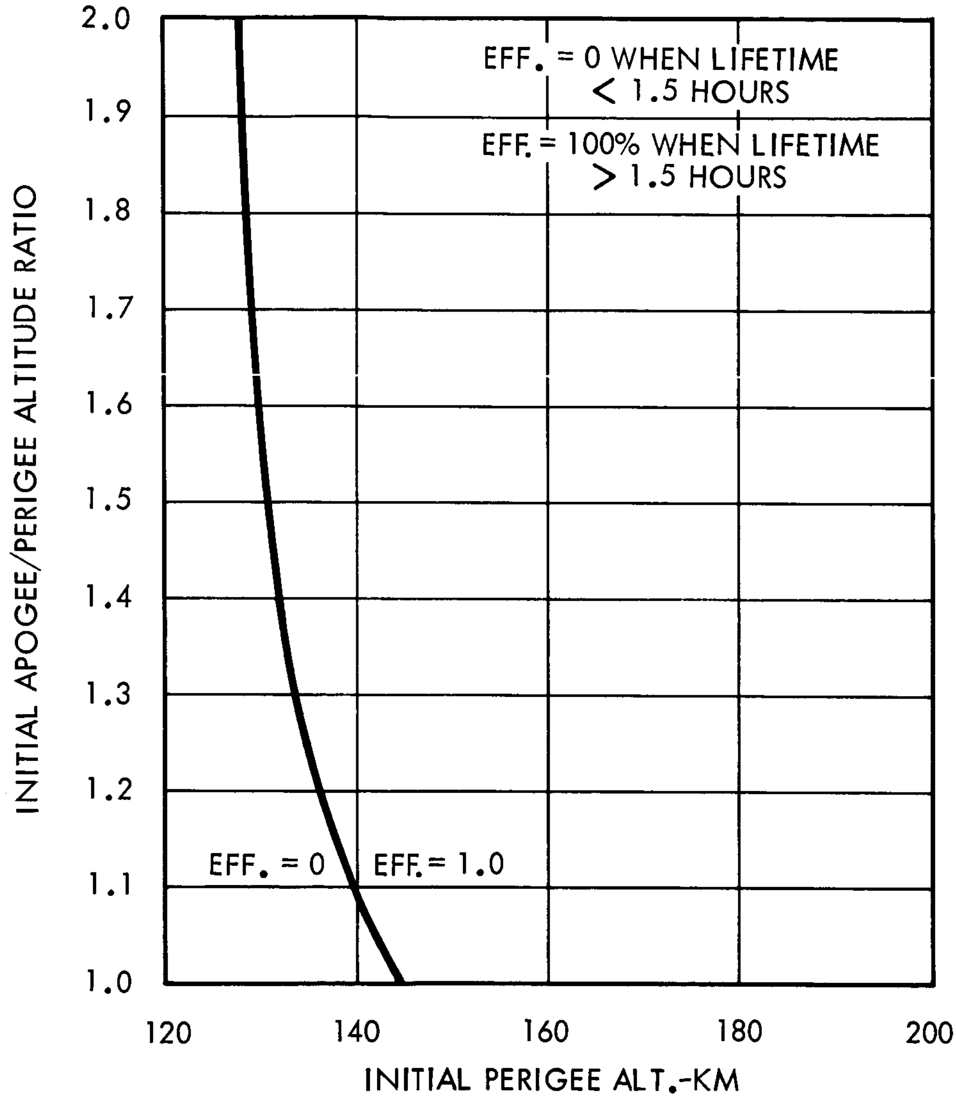


Figure B-9 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: SDT-5

Figure B-10 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: SDT-5

EXPERIMENT: SDT-5 **IBM PROBLEM NO.** 1439P020

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	3	(STEP FUNCTION)								
No. of Last Row	(IR)	2									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	1									

PERIGEE ALT. - km

EFFECTIVENESS

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	0.0	144.46	136.12	132.05	129.83	128.80	127.79								
2	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
3		APOGEE/PERIGEE ALT.													
4															
5															
6															
7															
8															
9															
10															
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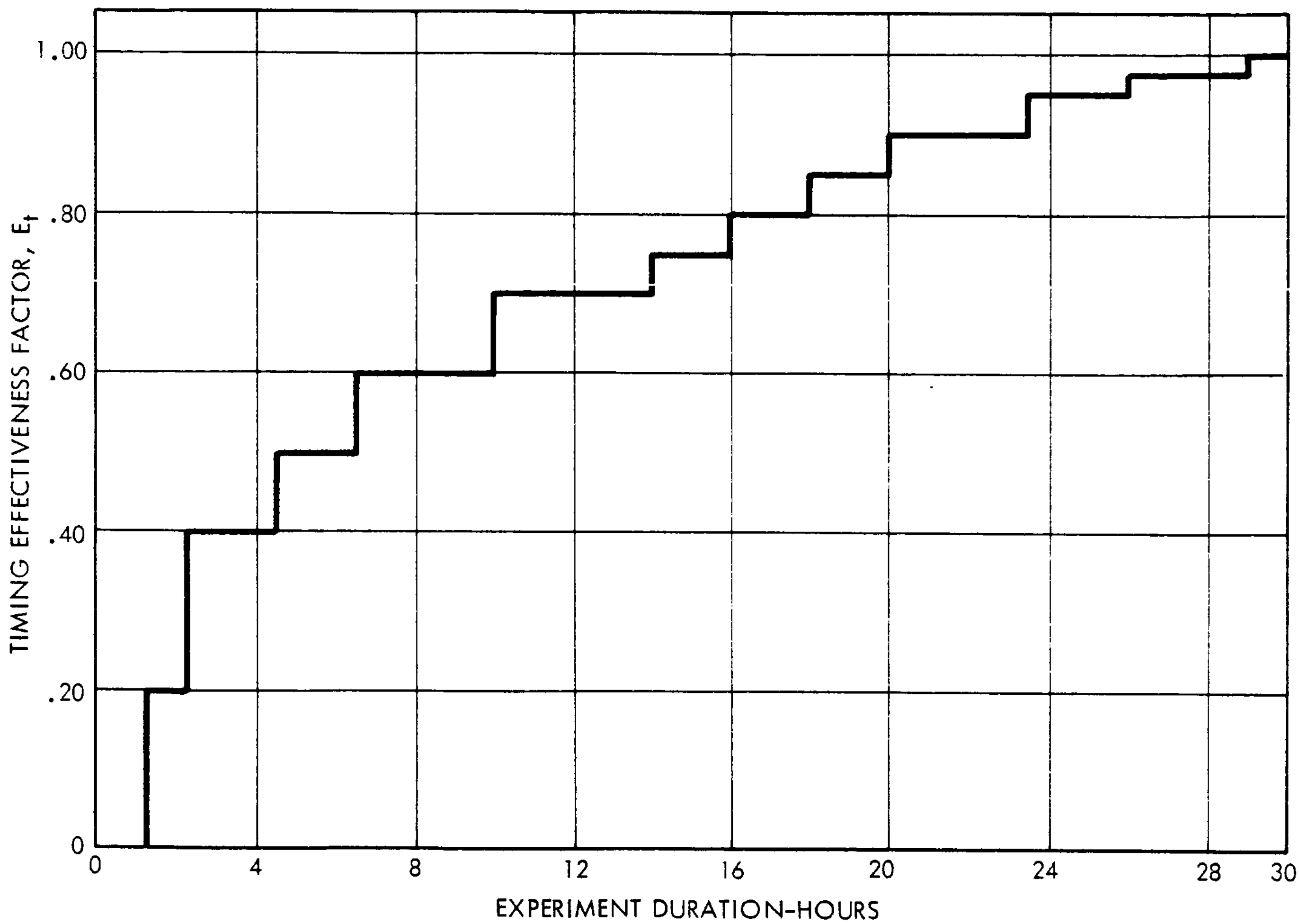

Figure B-11 BASIC EFFECTIVENESS DEFINITION - 1, EXPERIMENT: MS-3

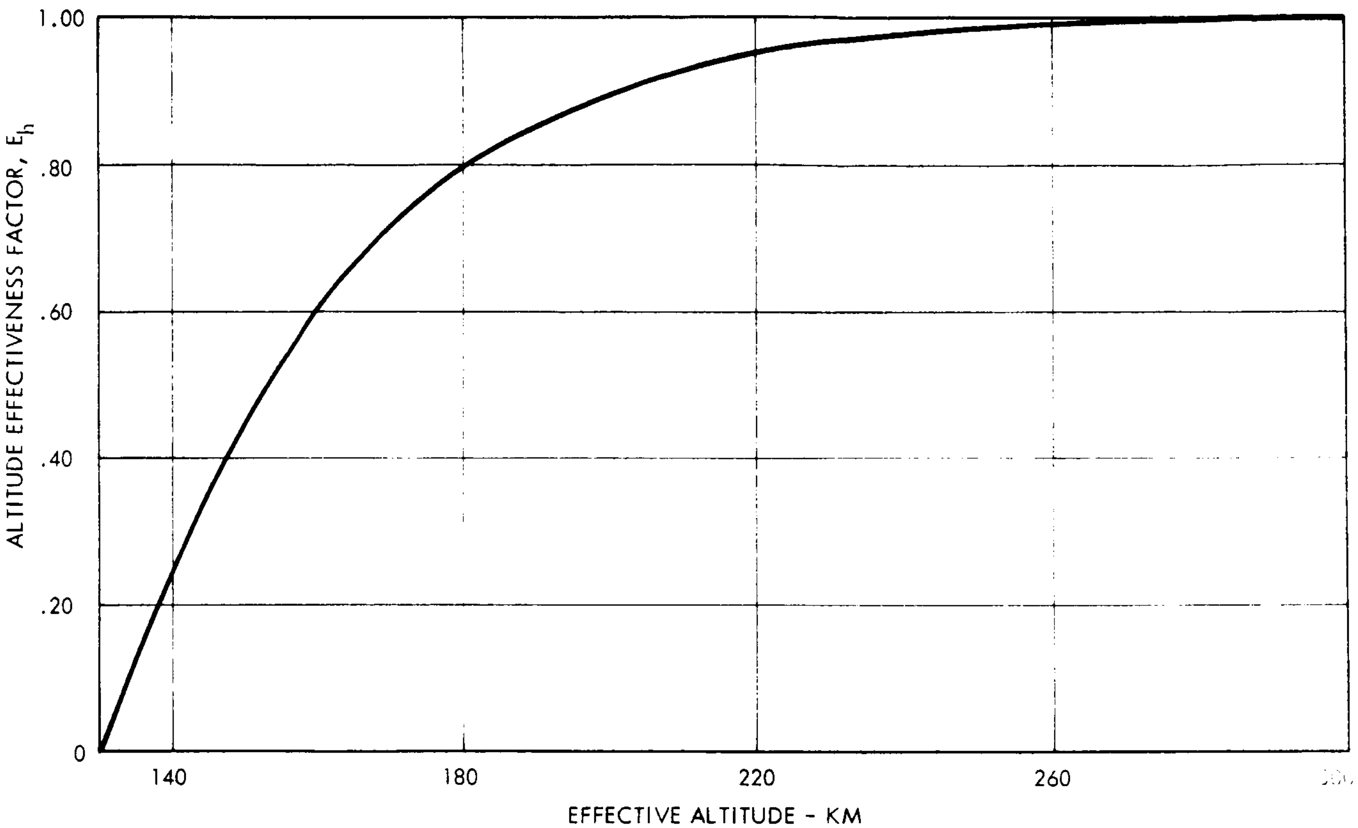

Figure B-12 BASIC EFFECTIVENESS DEFINITION - 2, EXPERIMENT: MS-3

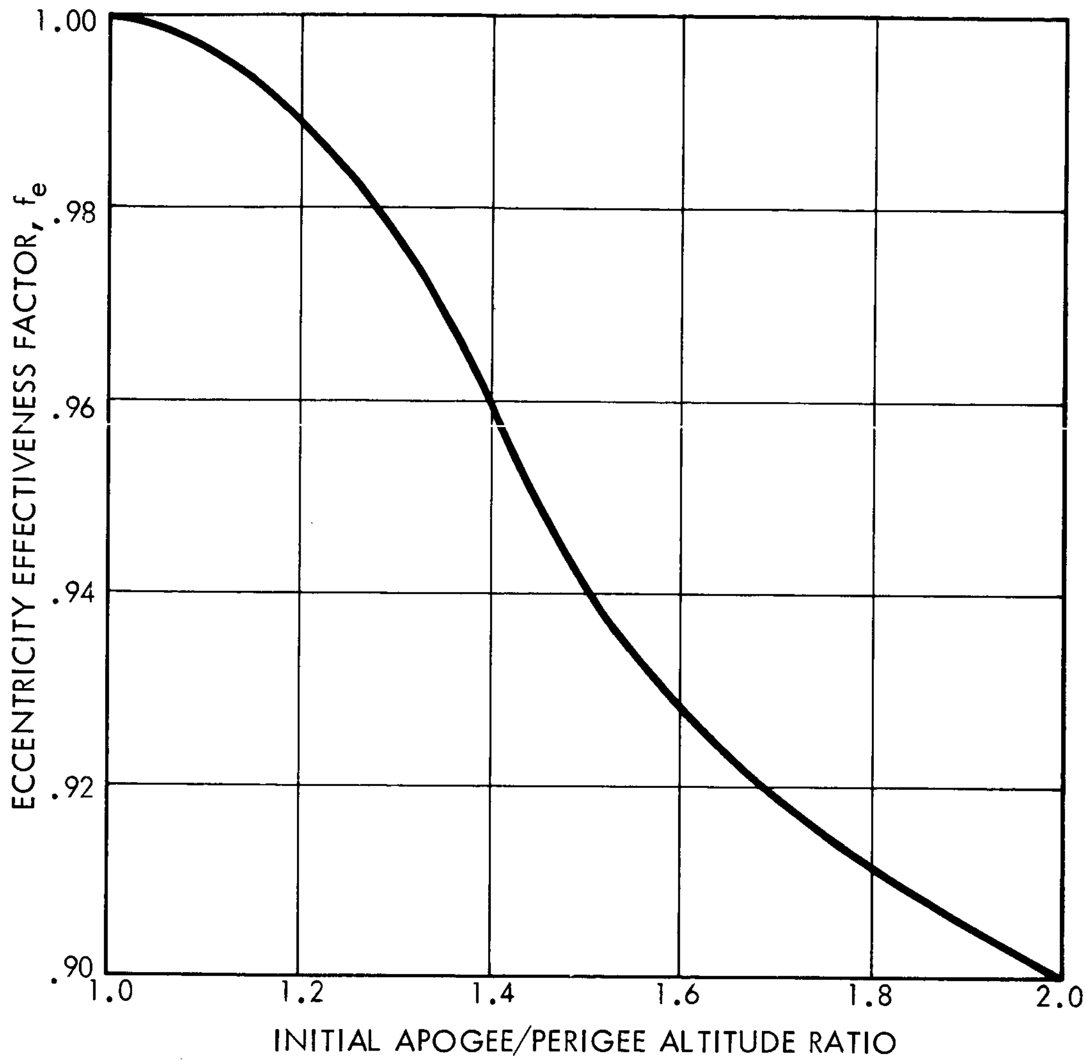

Figure B-13 BASIC EFFECTIVENESS DEFINITION - 3, EXPERIMENT: MS-3

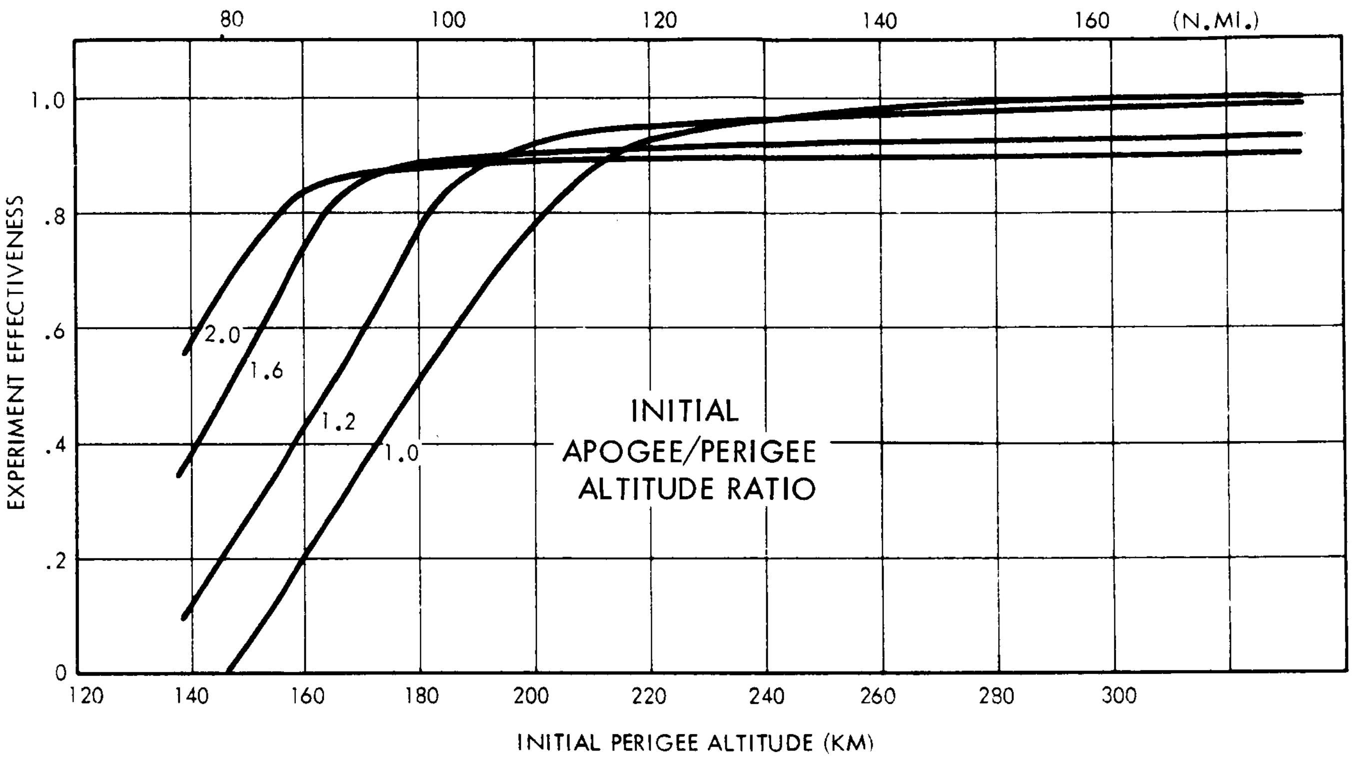

Figure B-14 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: MS-3

Figure B-15 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: MS-3

EXPERIMENT: MS-3 **IBM PROBLEM NO.** 1439P023

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	10									
Second Variable I.D.	(KY)	11									
Interp. Option	(KI)	2									
No. of Last Row	(IR)	5									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	7									
No. of Ordinate Values	(NY)	4									

EFFECTIVENESS

APOGEE/PERIGEE ALT. RATIO

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	2.0	0.71	0.864	0.88	0.89	0.893	0.897	0.90							
2	1.6	0.52	0.83	0.89	0.91	0.92	0.927	0.93							
3	1.2	0.24	0.53	0.84	0.95	0.968	0.98	0.99							
4	1.0	0.03	0.31	0.58	0.93	0.975	0.99	1.00							
5	0.0	148.16	166.68	185.2	222.24	259.28	296.32	333.36							
6			INITIAL PERIGEE ALT. - km												
7															
8															
9															
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

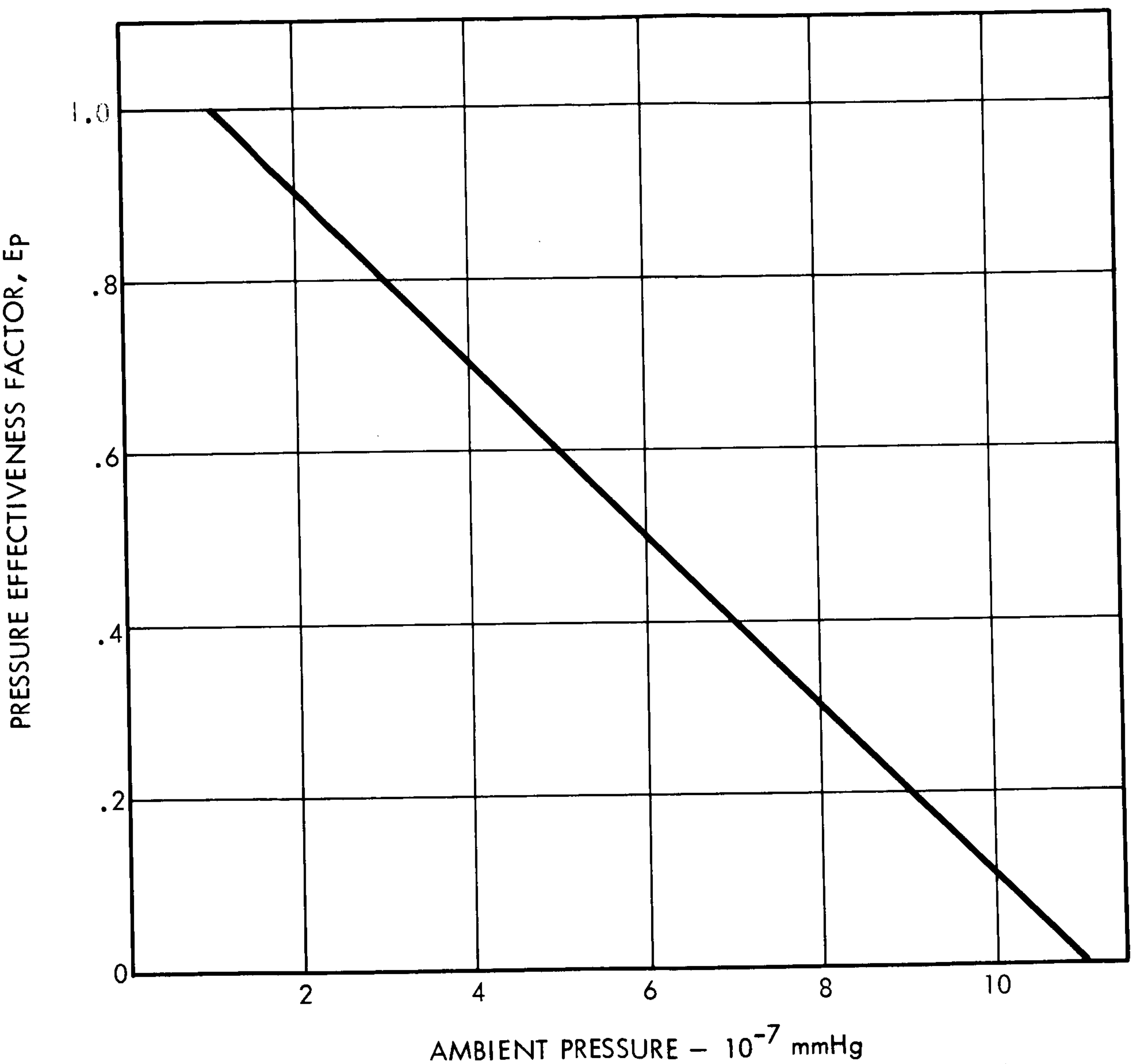

Figure B-16 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: MS-4

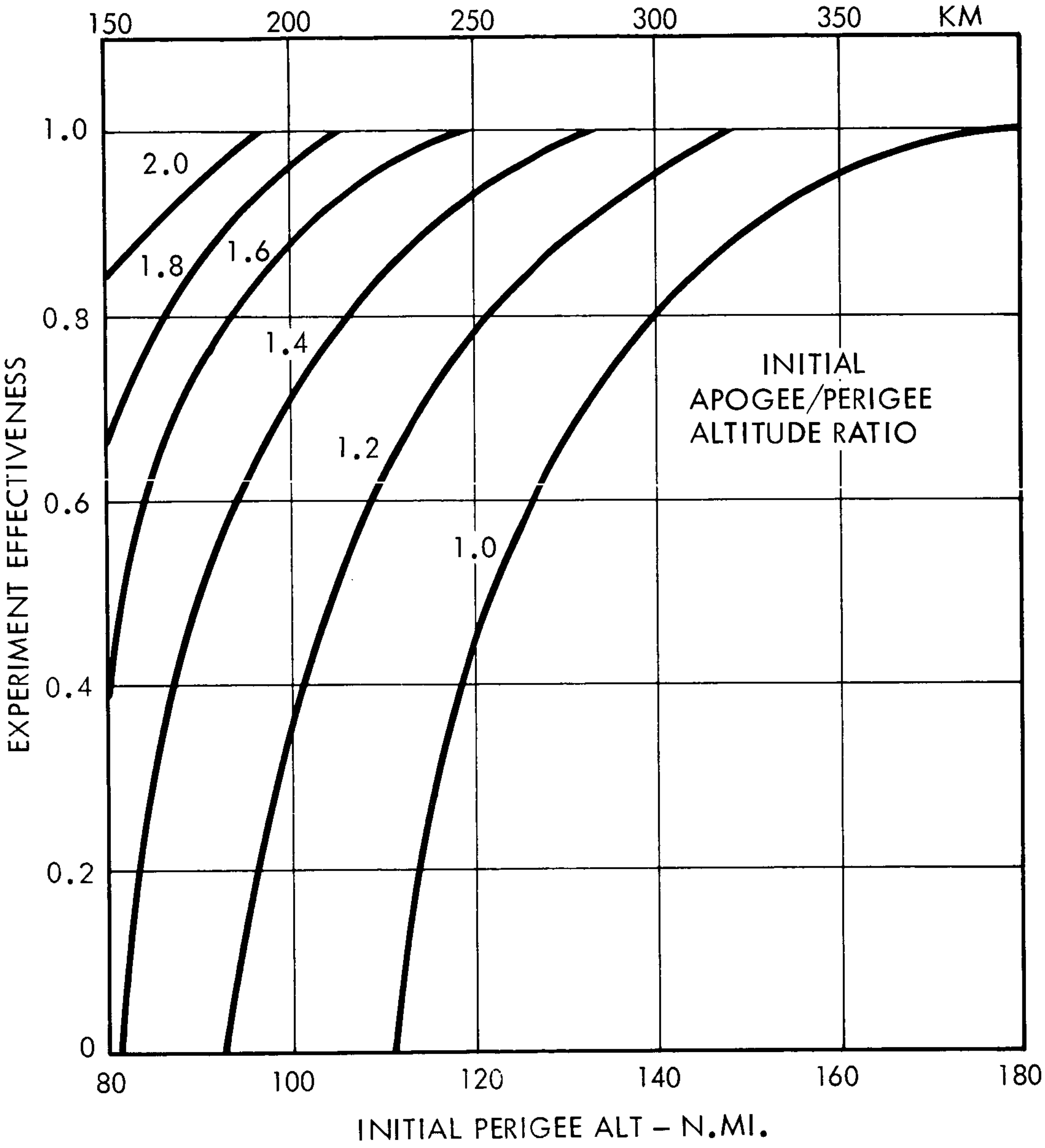

Figure B-17 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: MS-4

Figure B-18 EXPERIMENT EFFECTIVNESS LIBRARY WORK SHEET, EXPERIMENT: MS-4

EXPERIMENT: MS-4 **IBM PROBLEM NO.** 1439P024

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	10									
Second Variable I.D.	(KY)	11									
Interp. Option	(KI)	2									
No. of Last Row	(IR)	8									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	5									
No. of Ordinate Values	(NY)	7									

EFFECTIVENESS

APOGEE/PERIGEE ALT. RATIO

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	2.4	0.97	1.08	1.15	1.20	1.29									
2	2.0	0.84	1.02	1.11	1.17	1.25									
3	1.8	0.65	0.96	1.06	1.13	1.22									
4	1.6	0.30	0.875	1.00	1.08	1.18									
5	1.4	-1.00	0.71	0.93	1.02	1.13									
6	1.2	-10.00	0.35	0.79	0.95	1.07									
7	1.0	-100.00	-1.00	0.45	0.80	1.00									
8	0.0	148.16	185.2	222.24	259.28	333.36									
9		INITIAL PERIGEE ALT. - km													
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

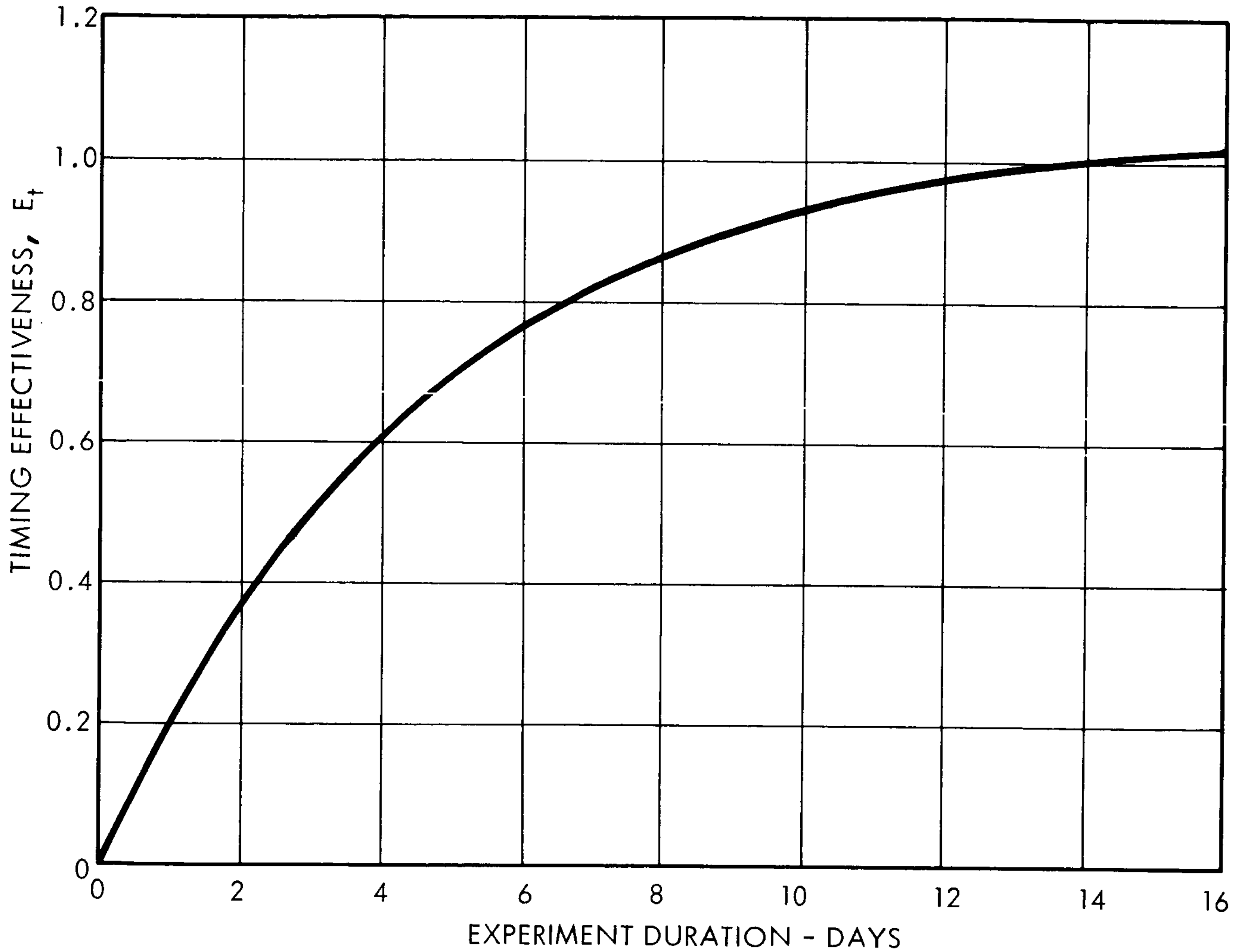

Figure B-19 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: MI-1, MI-2, OEA-5

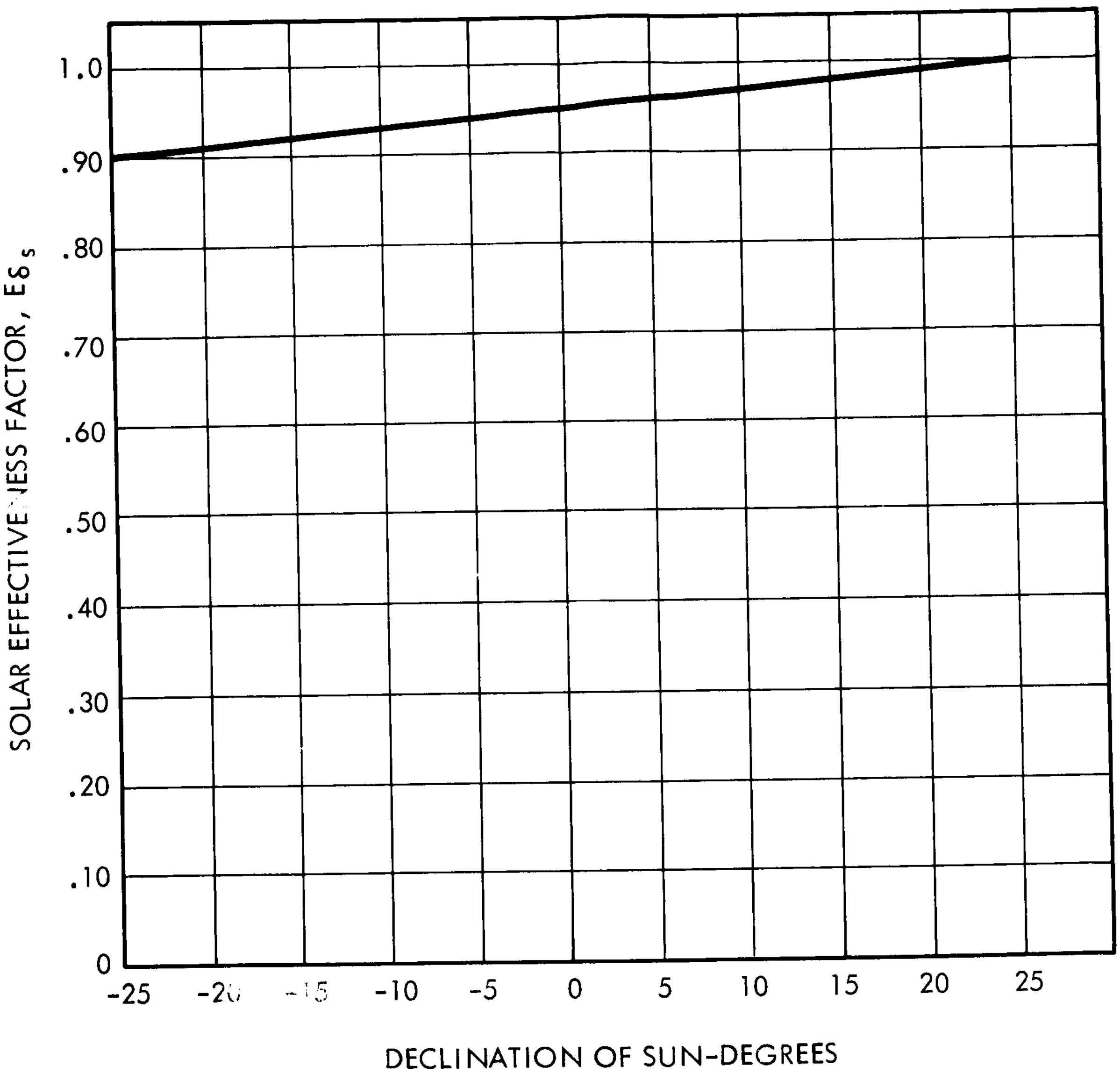

Figure B-20 FINAL EFFECTIVENESS DEFINITION - 1, EXPERIMENT: MI-1

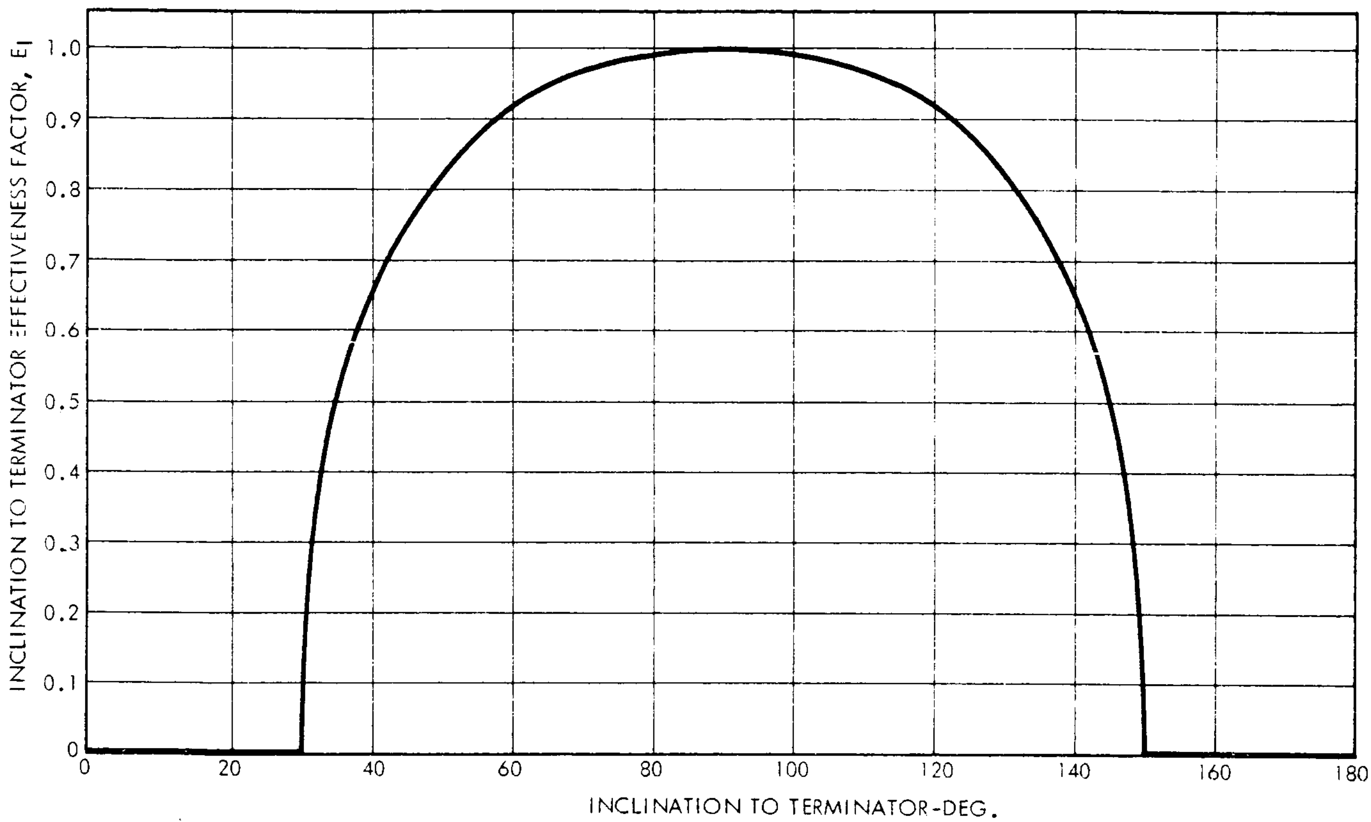

Figure B-21 FINAL EFFECTIVENESS DEFINITION - 2, EXPERIMENT: MI-1

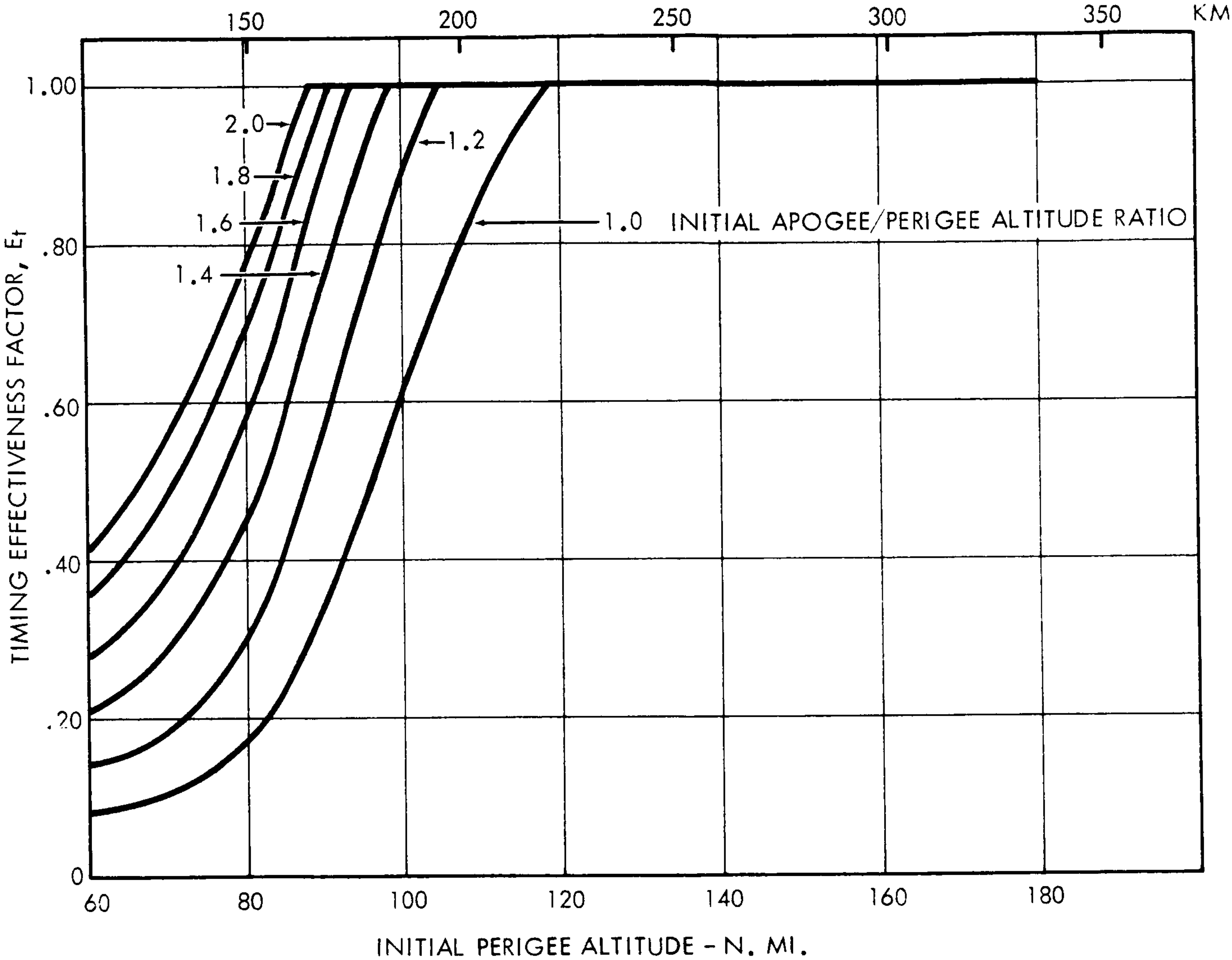

Figure B-22 FINAL EFFECTIVENESS DEFINITION - 3, EXPERIMENT: MI-1, MI-2

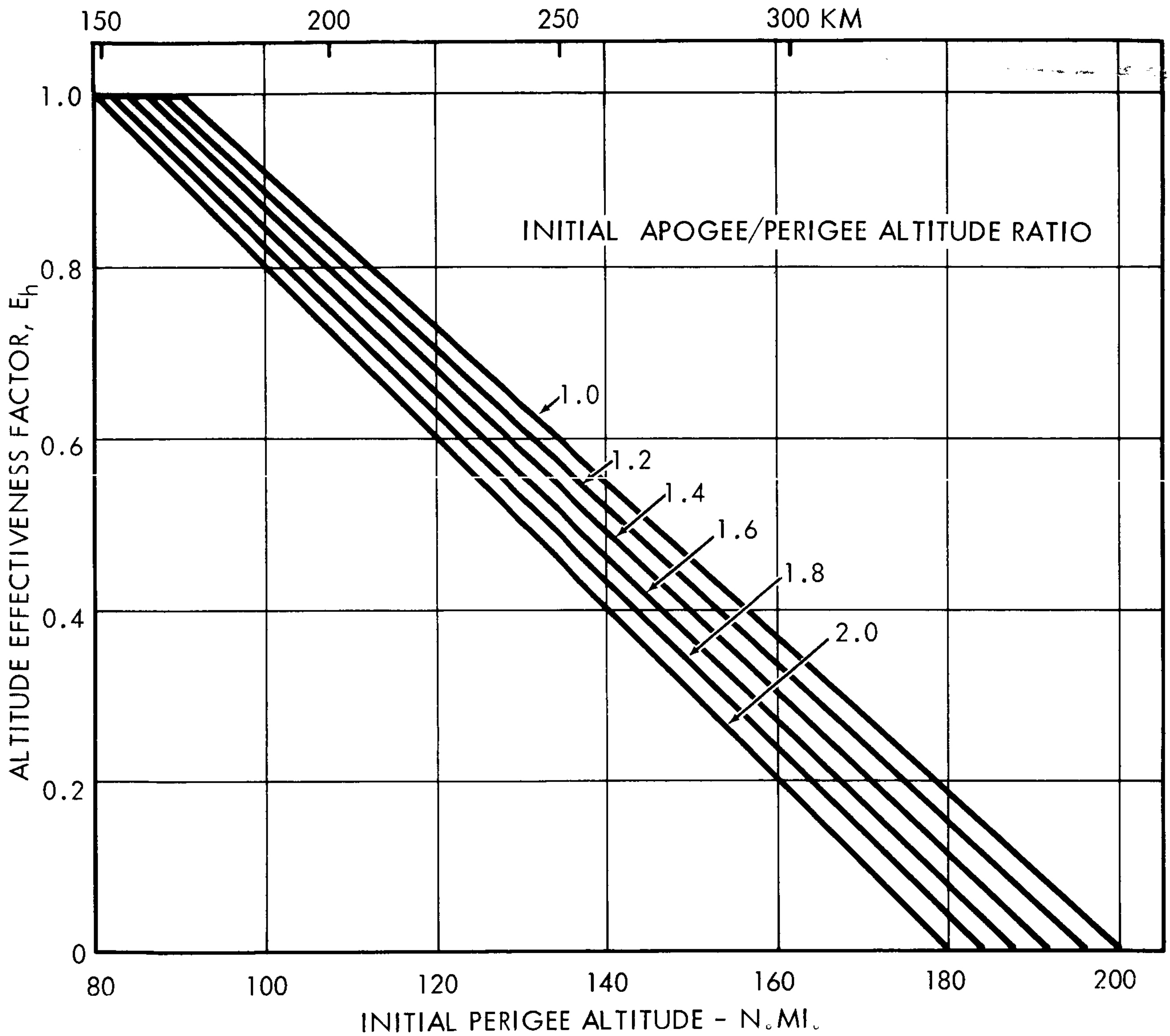

Figure B-23 FINAL EFFECTIVENESS DEFINITION - 4, EXPERIMENT: MI-1, MI-2

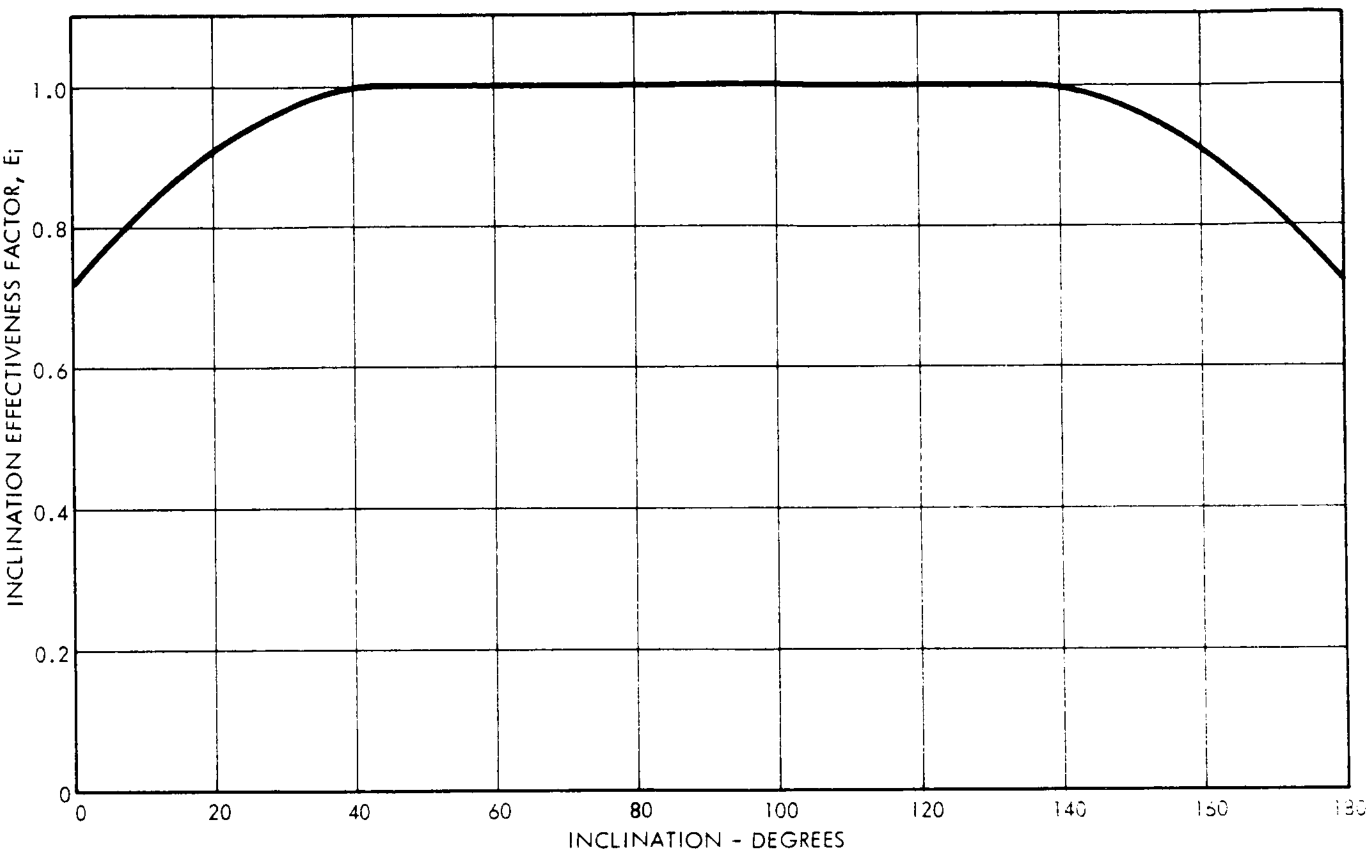

Figure B-24 FINAL EFFECTIVENESS DEFINITION - 5, EXPERIMENT: MI-1, MI-2

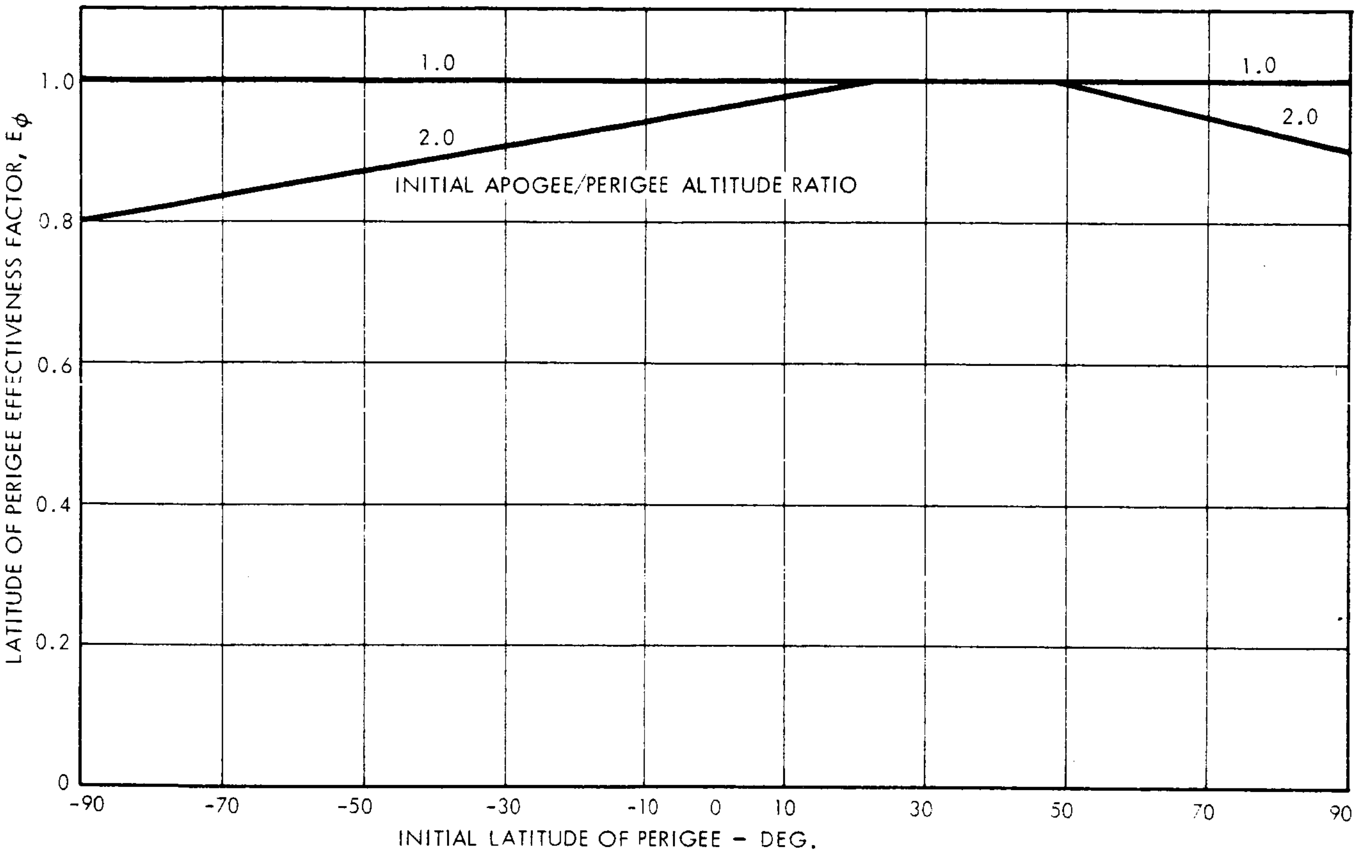

Figure B-25 FINAL EFFECTIVENESS DEFINITION - 6, EXPERIMENT: MI-1, MI-2

Figure B-26 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: MI-1

EXPERIMENT: MI-1 **IBM PROBLEM NO.** 1439P001

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	14	10	10	3	7	15				
Second Variable I.D.	(KY)	0	11	11	0	11	0				
Interp. Option	(KI)	1	1	1	1	1	1				
No. of Last Row	(IR)	2	11	20	24	7	11				
No. of First Column	(JC)	1	1	1	1	11	13				
No. of Abscissa Values	(NX)	14	6	8	14	4	2				
No. of Ordinate Values	(NY)	1	6	6	1	2	1				

APOGEE/PERIGEE ALT. (rows 1–11); APOGEE/PERIGEE ALT. (rows 14–20); APOGEE/PERIGEE ALT. RATIO (Table No. 5)

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
	TABLE NO. 1		E_I												
1	0.0	0.0	0.0	0.53	0.70	0.82	0.92	0.97	1.0	0.97	0.91	0.77	0.53	0.0	0.0
2	0.0	0.0	30.0	36.0	42.0	50.0	60.0	70.0	90.0	110.0	122.0	134.0	144.0	150.0	180.0
3					INCLINATION TO TERMINATOR - DEGREES										
4	TABLE NO. 2		E_t								TABLE NO. 5		$E_\emptyset$		
5	2.0	0.41	0.77	1.01	1.05	1.05	1.05				2.0	0.80	1.0	1.0	0.90
6	1.8	0.35	0.69	0.97	1.05	1.05	1.05				1.0	1.0	1.0	1.0	1.0
7	1.6	0.28	0.57	0.89	1.05	1.05	1.05				0.0	-90.	23.	48.	90.
8	1.4	0.21	0.45	0.76	1.03	1.05	1.05					LATITUDE OF PERIGEE-DEGREES	TABLE NO. 6		E_δ
9	1.2	0.14	0.30	0.57	0.88	1.05	1.05						0.0	0.90	1.0
10	1.0	0.08	0.17	0.35	0.59	1.01	1.05						0.0	-25.	25.
11	0.0	111.12	148.16	166.68	185.2	222.24	333.36						DECLINATION OF SUN-DEGREES		
12		PERIGEE ALTITUDE - km													
13	TABLE NO. 3		E_h												
14	2.0	1.20	1.00	0.90	0.80	0.60	0.40	0.20	0.0						
15	1.8	1.22	1.02	0.92	0.82	0.63	0.43	0.23	0.04						
16	1.6	1.24	1.04	0.94	0.85	0.66	0.46	0.27	0.08						
17	1.4	1.25	1.06	0.96	0.87	0.68	0.49	0.30	0.11						
18	1.2	1.27	1.07	0.98	0.89	0.70	0.52	0.33	0.15						
19	1.0	1.25	1.09	1.00	0.91	0.73	0.54	0.37	0.18						
20	0.0	111.12	148.16	166.68	185.20	222.24	259.28	296.32	333.36						
21		PERIGEE ALTITUDE - km													
22	TABLE NO. 4		E_i												
23	0.0	0.72	0.82	0.91	0.95	0.98	0.99	1.0	1.0	0.99	0.98	0.95	0.91	0.82	0.72
24	0.0	0.0	10.0	20.0	27.0	34.0	38.0	48.0	132.0	142.0	146.0	153.0	160.0	170.0	180.0
25			INCLINATIONS - DEGREES												

Figure B-27 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: MI-2

EXPERIMENT: MI-2 **IBM PROBLEM NO.** 1439P002

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	3	10	10	7						
Second Variable I.D.	(KY)	0	11	11	11						
Interp. Option	(KI)	1	1	1	1						
No. of Last Row	(IR)	2	11	20	25						
No. of First Column	(JC)	1	1	1	1						
No. of Abscissa Values	(NX)	14	6	8	4						
No. of Ordinate Values	(NY)	1	6	6	2						

		1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
		TABLE NO. 1		E_i												
APOGEE/PERIGEE ALT.	1	0.0	0.72	0.82	0.91	0.95	0.98	0.99	1.0	1.0	0.99	0.98	0.95	0.91	0.82	0.72
	2	0.0	0.0	10.0	20.0	27.0	34.0	38.0	48.0	132.0	142.0	146.0	153.0	160.0	170.0	180.0
	3			INCLINATION - DEGREES												
	4	TABLE NO. 2		E_t												
	5	2.0	0.41	0.77	1.01	1.05	1.05	1.05								
	6	1.8	0.35	0.69	0.97	1.05	1.05	1.05								
	7	1.6	0.28	0.57	0.89	1.05	1.05	1.05								
	8	1.4	0.21	0.45	0.76	1.03	1.05	1.05								
	9	1.2	0.14	0.30	0.57	0.88	1.05	1.05								
	10	1.0	0.08	0.17	0.35	0.59	1.01	1.05								
	11	0.0	111.12	148.16	166.68	185.2	222.24	333.36								
	12		PERIGEE ALTITUDE - km													
APOGEE/PERIGEE ALT.	13	TABLE NO. 3		E_h												
	14	2.0	1.20	1.00	0.90	0.80	0.60	0.40	0.20	0.00						
	15	1.8	1.22	1.02	0.92	0.82	0.63	0.43	0.23	0.04						
	16	1.6	1.24	1.04	0.94	0.85	0.66	0.46	0.27	0.08						
	17	1.4	1.25	1.06	0.96	0.87	0.68	0.49	0.30	0.11						
	18	1.2	1.27	1.07	0.98	0.89	0.70	0.52	0.33	0.15						
	19	1.0	1.29	1.09	1.00	0.91	0.73	0.54	0.37	0.18						
	20	0.0	111.12	148.16	166.68	185.20	222.24	259.28	296.32	333.36						
	21		PERIGEE ALTITUDE - km													
APOG/PERIG. ALT.	22	TABLE NO. 4		$E_\emptyset$												
	23	2.0	0.80	1.00	1.00	0.90										
	24	1.0	1.00	1.00	1.00	1.00										
	25	0.0	-90.0	23.0	48.0	90.0	LATITUDE OF PERIGEE - DEGREES									

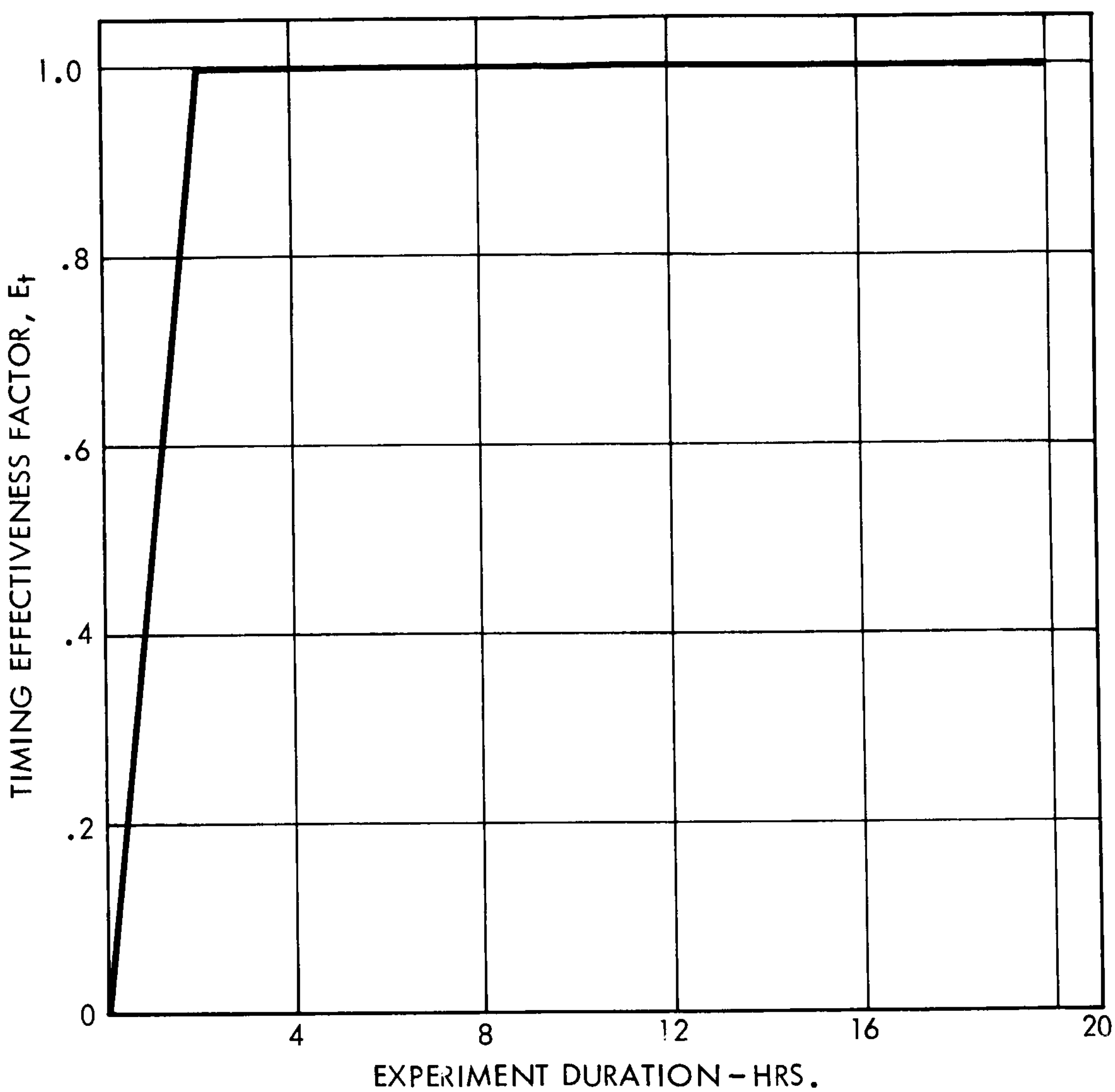

Figure B-28 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: SLG-1, SLG-4

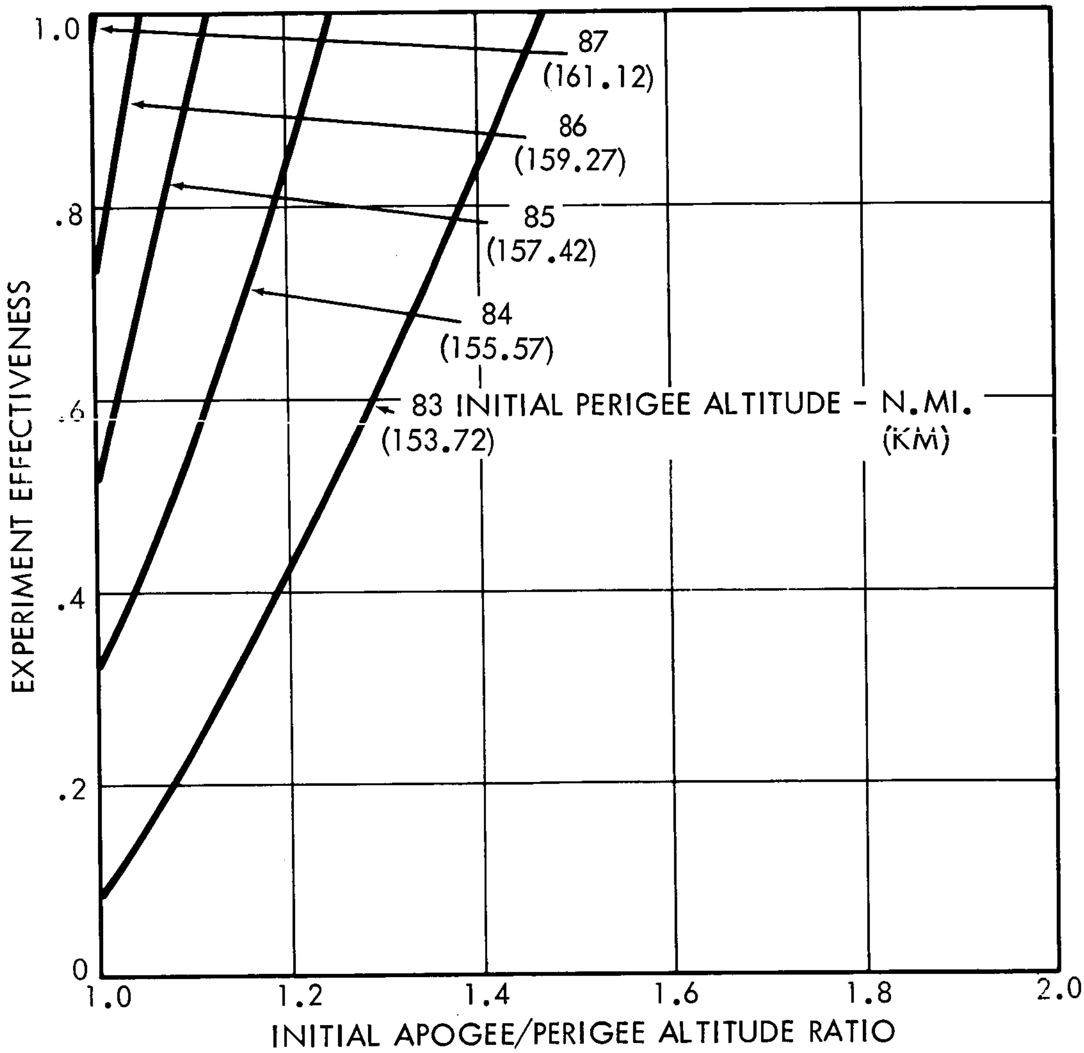

Figure B-29 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: SLG-1

Figure B-30 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: SLG-1

EXPERIMENT: SLG-1 **IBM PROBLEM NO.** 1439P006

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	1									
No. of Last Row	(IR)	10									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	9									

EFFECTIVENESS

PERIGEE ALT. - km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	333.36	1.0	1.0	1.0	1.0	1.0	1.0								
2	162.98	1.22	2.71	1.0	1.0	1.0	1.0								
3	161.12	0.97	2.12	1.0	1.0	1.0	1.0								
4	159.27	0.73	1.64	1.0	1.0	1.0	1.0								
5	157.42	0.51	1.16	3.0	1.0	1.0	1.0								
6	155.57	0.32	0.74	2.0	1.0	1.0	1.0								
7	153.72	0.15	0.35	1.0	1.0	1.0	1.0								
8	151.86	0.0	0.0	0.0	0.0	0.0	0.0								
9	111.12	0.0	0.0	0.0	0.0	0.0	0.0								
10	0.0	1.0	1.16	1.47	1.6	1.8	2.0								
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

APOGEE/PERIGEE ALT. RATIO

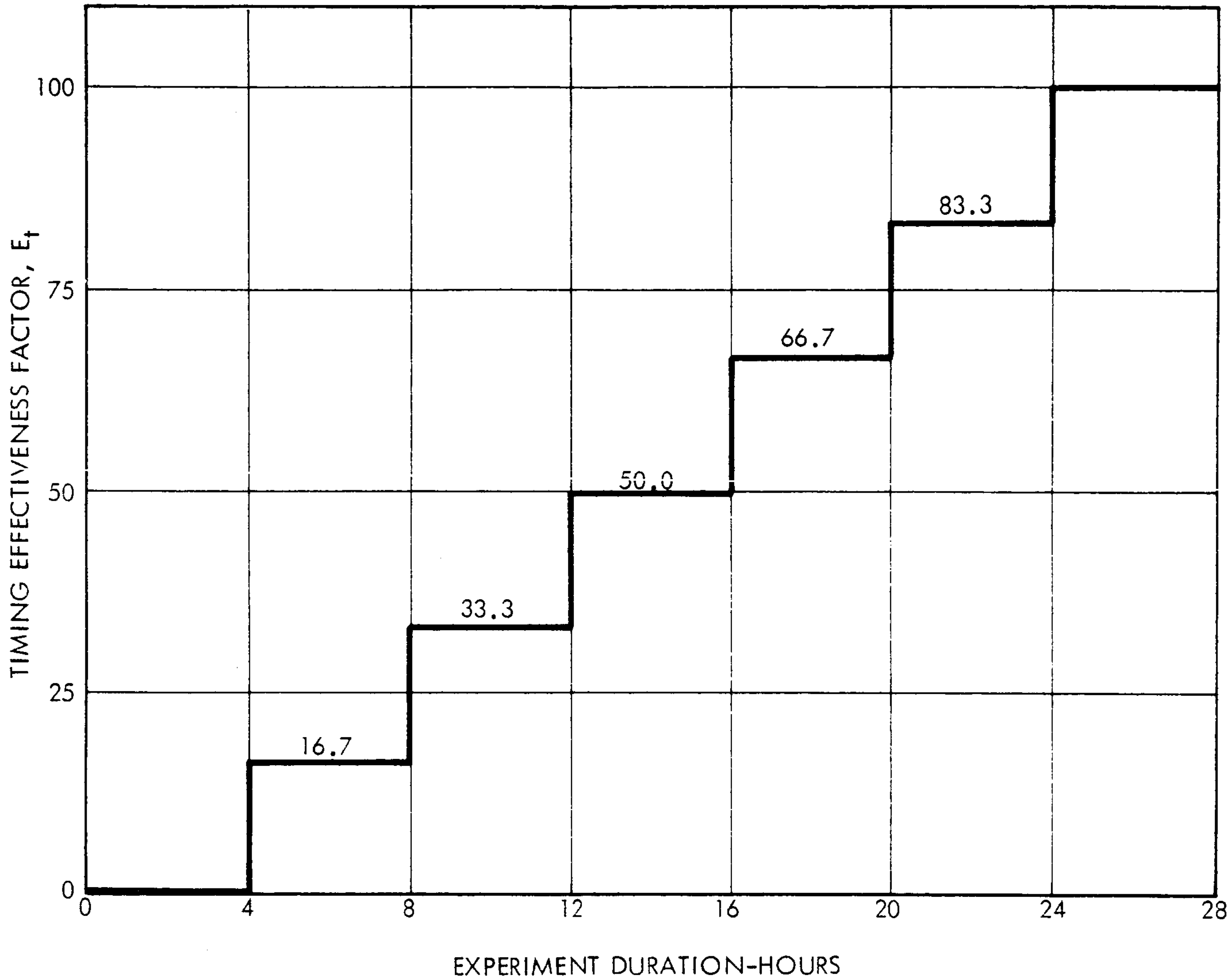

Figure B-31 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: SLG-2

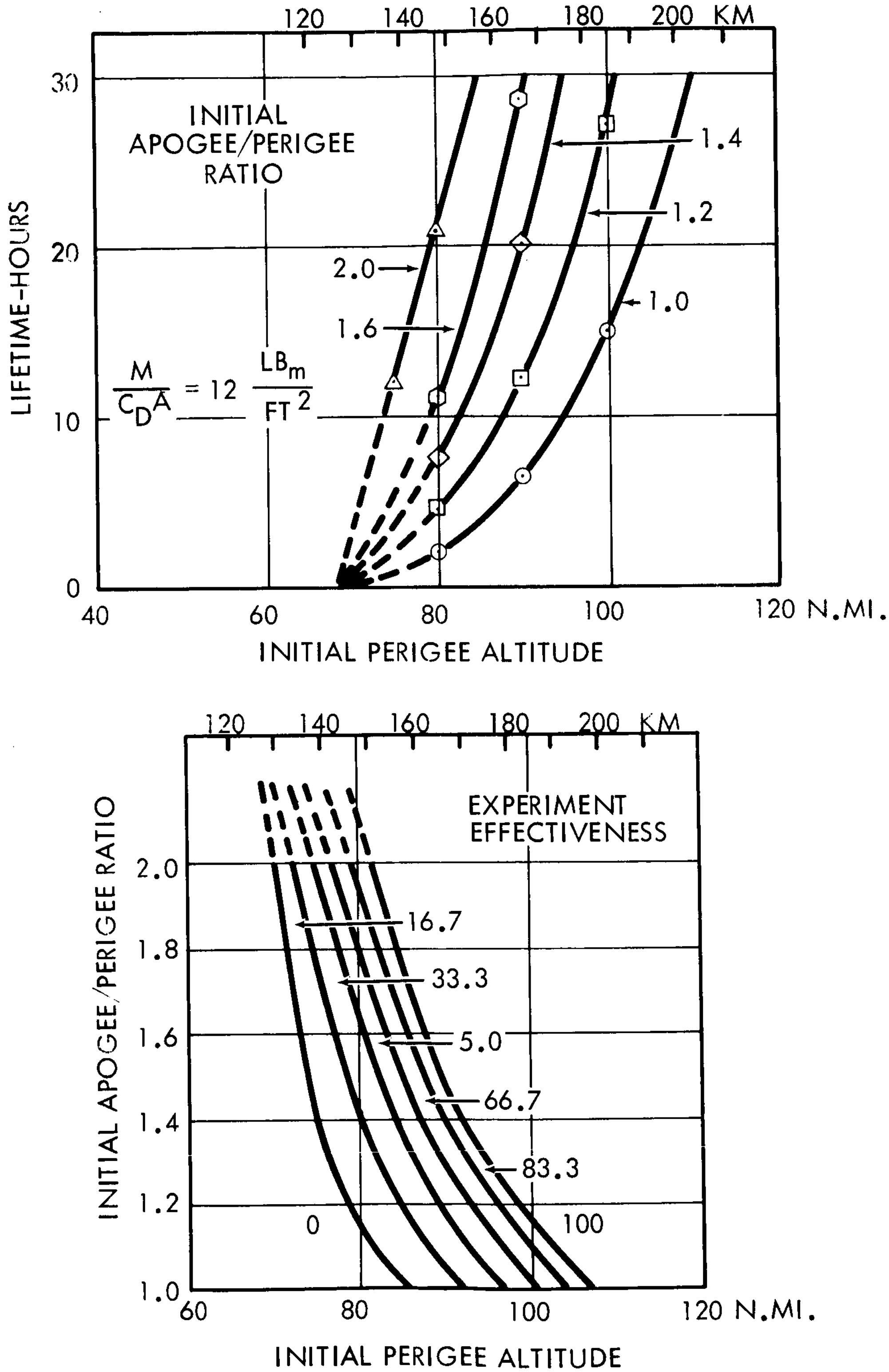

Figure B-32 FINAL EFFECITIVENESS DEFINITIONS, EXPERIMENT: SLG-2

Figure B-33 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: SLG-2

EXPERIMENT: SLG-2 **IBM PROBLEM NO.** 1439P007

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	4	(STEP FUNCTION)								
No. of Last Row	(IR)	7									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	7									
No. of Ordinate Values	(NY)	6									

EFFECTIVENESS

INITIAL PERIGEE ALT. - km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	0.833	198.16	182.42	170.38	162.61	157.23	151.49	146.31							
2	0.667	192.61	177.79	166.38	158.90	152.60	147.23	141.31							
3	0.500	186.68	172.24	161.68	154.64	148.72	142.79	137.42							
4	0.333	179.64	165.75	155.94	150.01	144.46	138.90	133.71							
5	0.167	170.38	157.42	148.90	143.34	138.90	134.27	129.83							
6	0.0	159.27	146.31	139.46	135.57	133.34	130.38	127.79							
7	0.0	1.0	1.2	1.4	1.6	1.8	2.0	2.2							
8			APOGEE/PERIGEE ALT. RATIO												
9															
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
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24															
25															

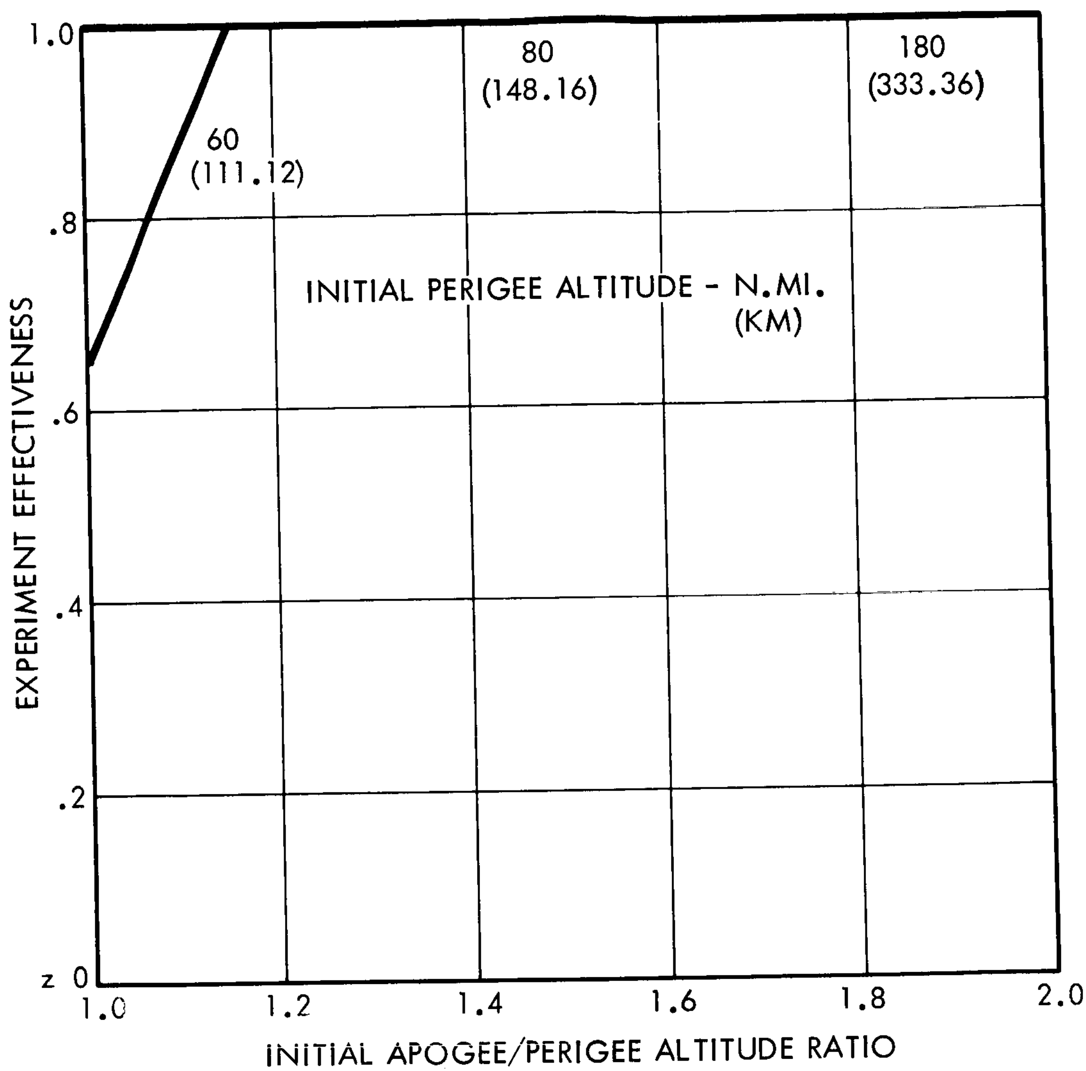

Figure B-34 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: SLG-4

Figure B-35 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: SLG-4

EXPERIMENT: SLG-4 **IBM PROBLEM NO.** 1439P009

		TABLE NO.									
		1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	1									
No. of Last Row	(IR)	4									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	3									

EFFECTIVENESS

PERIGEE ALT. – km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	333.36	3.0	6.0	10.0	15.0	20.0	25.0								
2	148.16	1.38	2.6	4.2	6.0	8.2	10.0								
3	111.12	0.64	1.10	1.74	2.38	3.00	3.72								
4	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
5			APOGEE/PERIGEE ALT. RATIO												
6															
7															
8															
9															
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

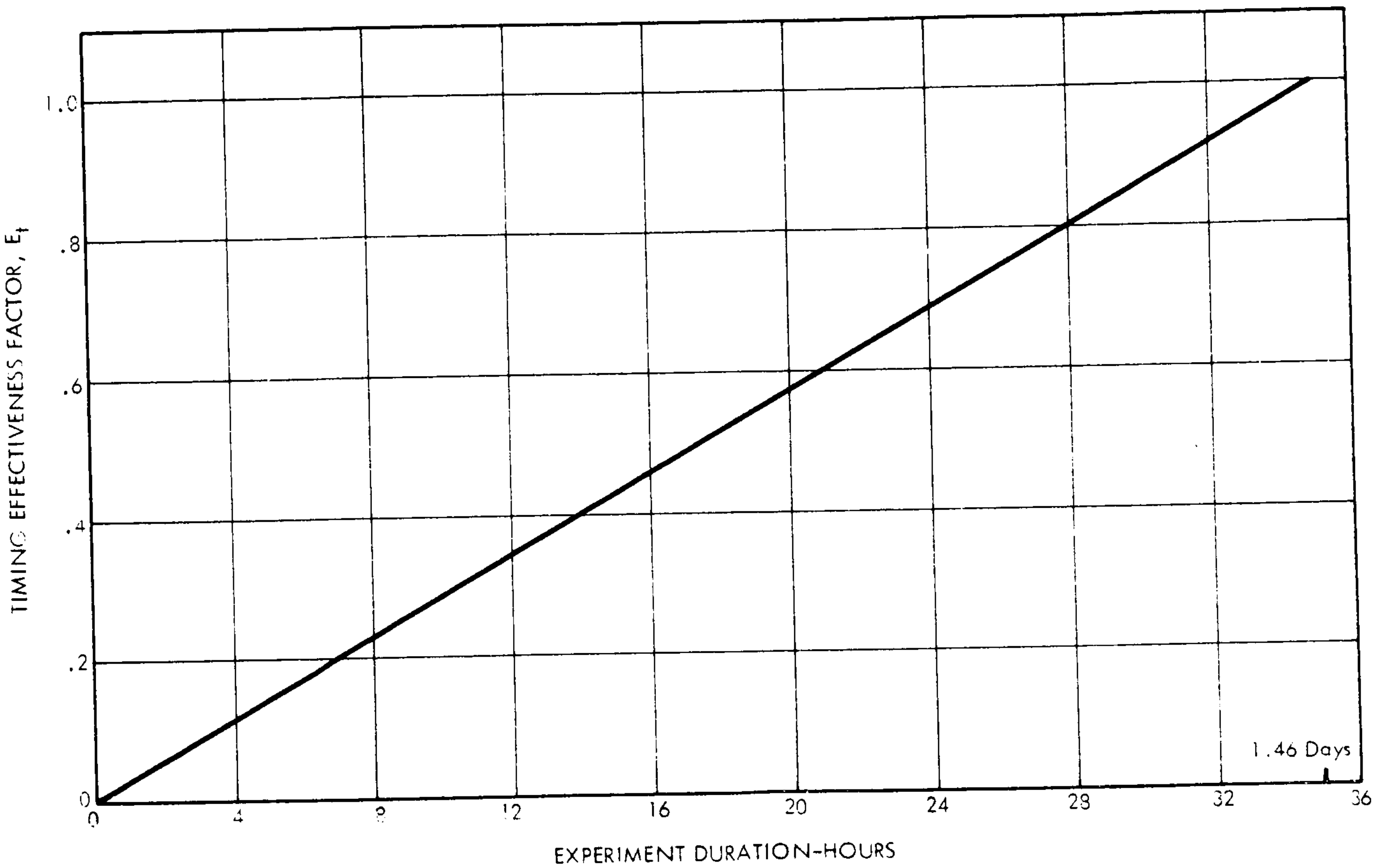

Figure B-36 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: SLG-5

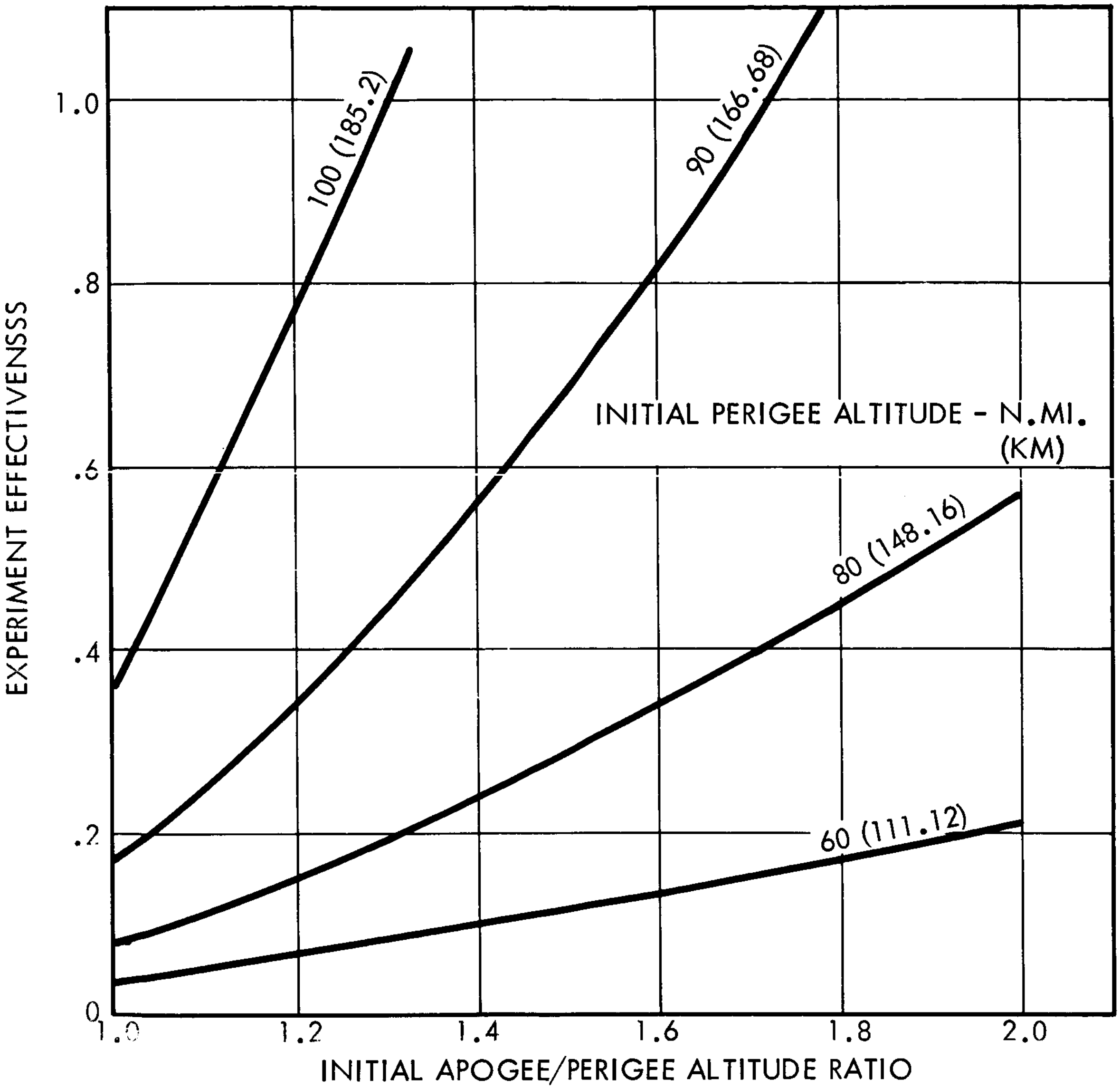

Figure B-37 FINAL EXPERIMENT DEFINITION, EXPERIMENT: SLG-5

Figure B-38 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: SLG-5

EXPERIMENT: SLG-5 **IBM PROBLEM NO.** 1439P010

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	1									
No. of Last Row	(IR)	7									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	6									

EFFECTIVENESS

PERIGEE ALT. - km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	336.36	3.9	7.6	4.8	6.8	2.6	3.4								
2	222.24	1.42	3.01	4.73	6.71	2.6	3.4								
3	185.20	0.36	0.77	1.21	1.85	2.55	3.36								
4	166.68	0.17	0.34	0.56	0.81	1.12	1.44								
5	148.16	0.079	0.15	0.24	0.34	0.47	0.57								
6	111.12	0.036	0.063	0.099	0.136	0.17	0.21								
7	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
8			APOGEE/PERIGEE ALT. RATIO												
9															
10															
11															
12															
13															
14															
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25															

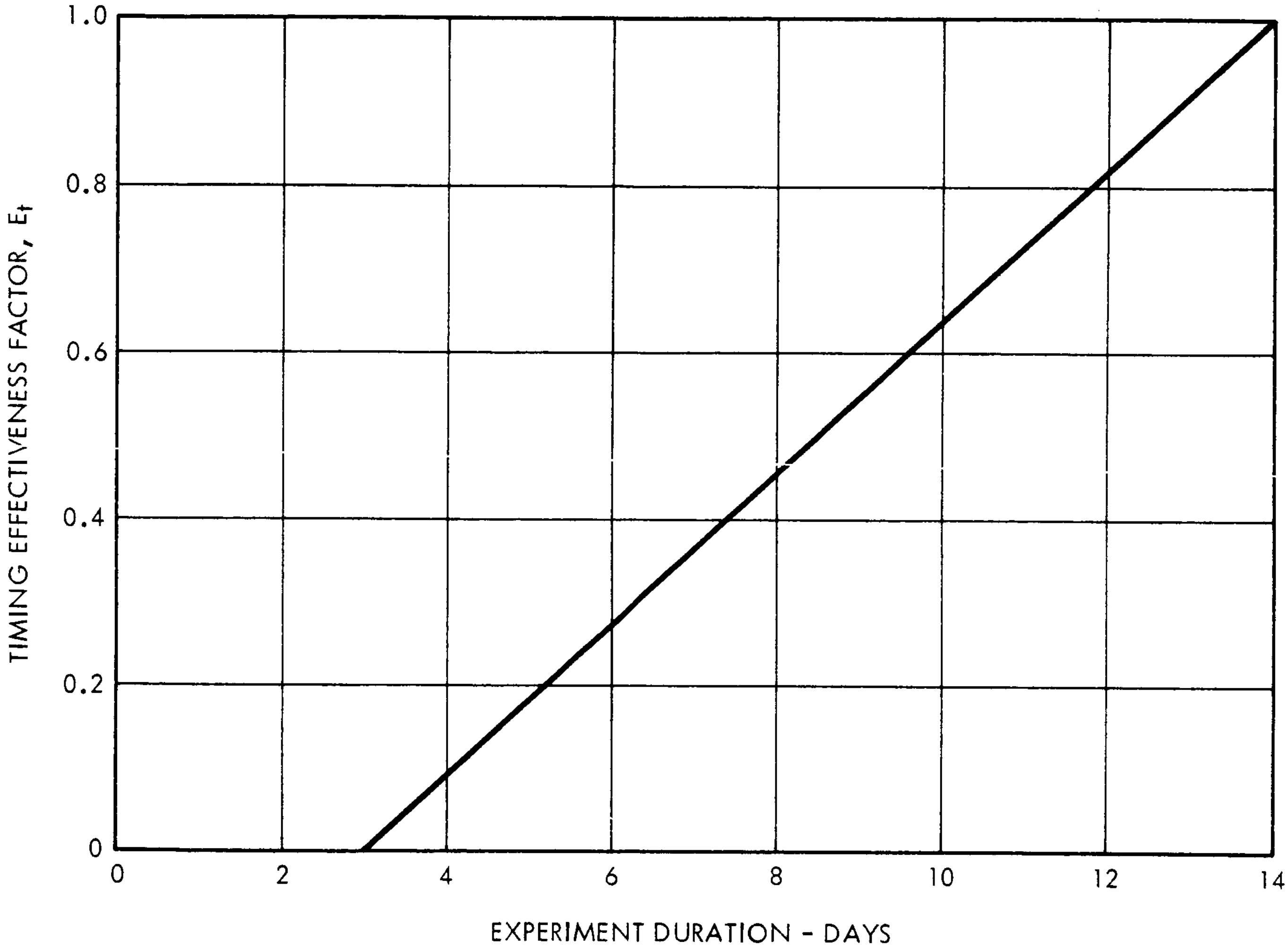

Figure B-39 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: M-1

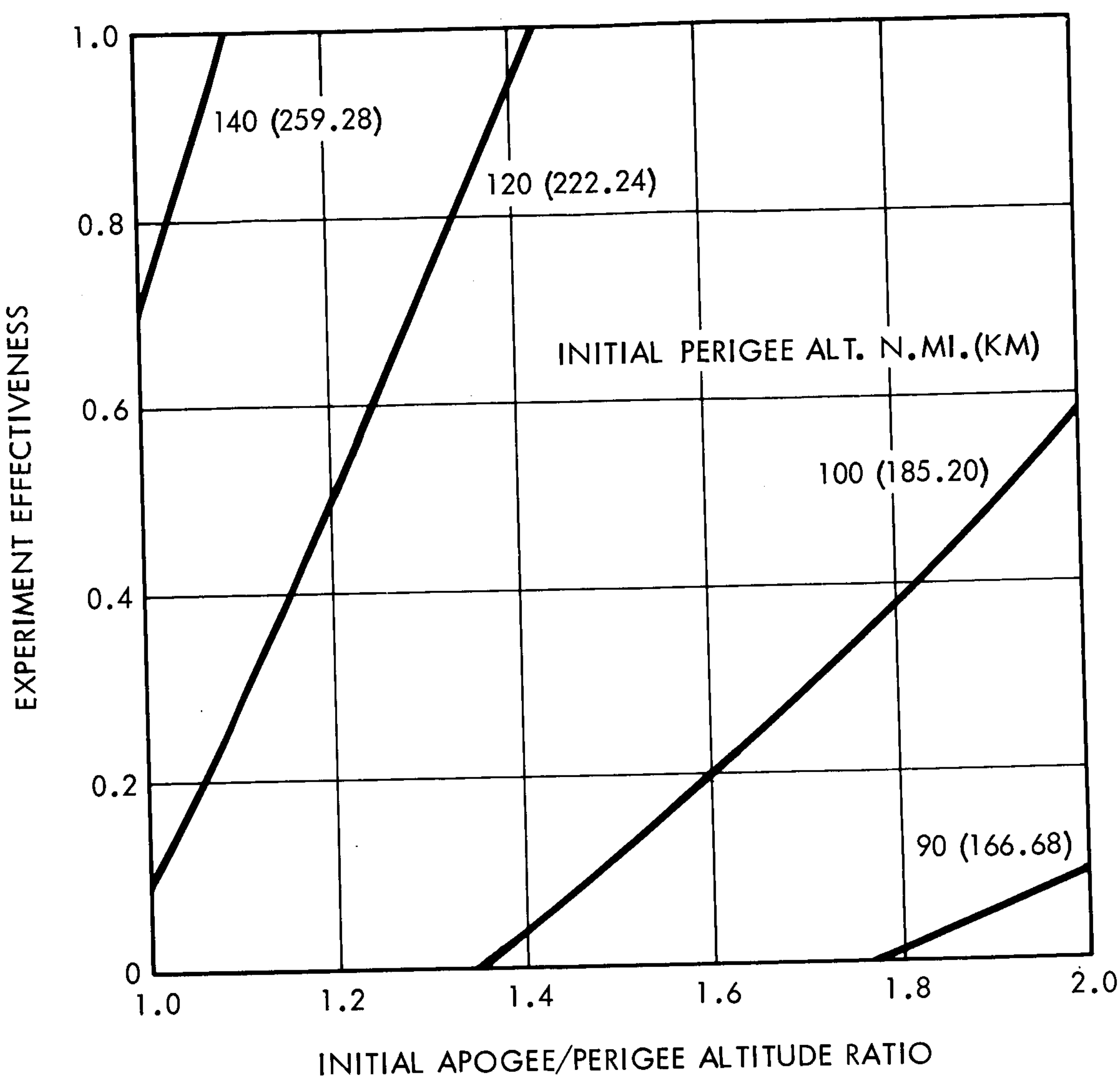

Figure B-40 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: M-1

Figure B-41 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: M-1

EXPERIMENT: M-1 **IBM PROBLEM NO.** 1439P011

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	2									
No. of Last Row	(IR)	9									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	8									

PERIGEE ALT. - km

EFFECTIVENESS

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	333.36	4.44	8.18	14.4	22.7	34.5	48.5								
2	296.32	1.82	3.74	6.10	7.18	12.5	16.5								
3	259.28	0.700	1.66	2.52	3.89	5.25	6.78								
4	222.24	0.090	0.494	0.930	1.44	1.94	2.52								
5	185.20	-0.18	-0.075	0.035	0.198	0.375	0.581								
6	166.68	-0.24	-0.19	-0.13	-0.067	0.010	0.093								
7	148.16	-0.25	-0.24	-0.21	-0.19	-0.15	-0.13								
8	111.12	-0.26	-0.26	-0.25	-0.24	-0.23	-0.23								
9	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

APOGEE/PERIGEE ALT. RATIO

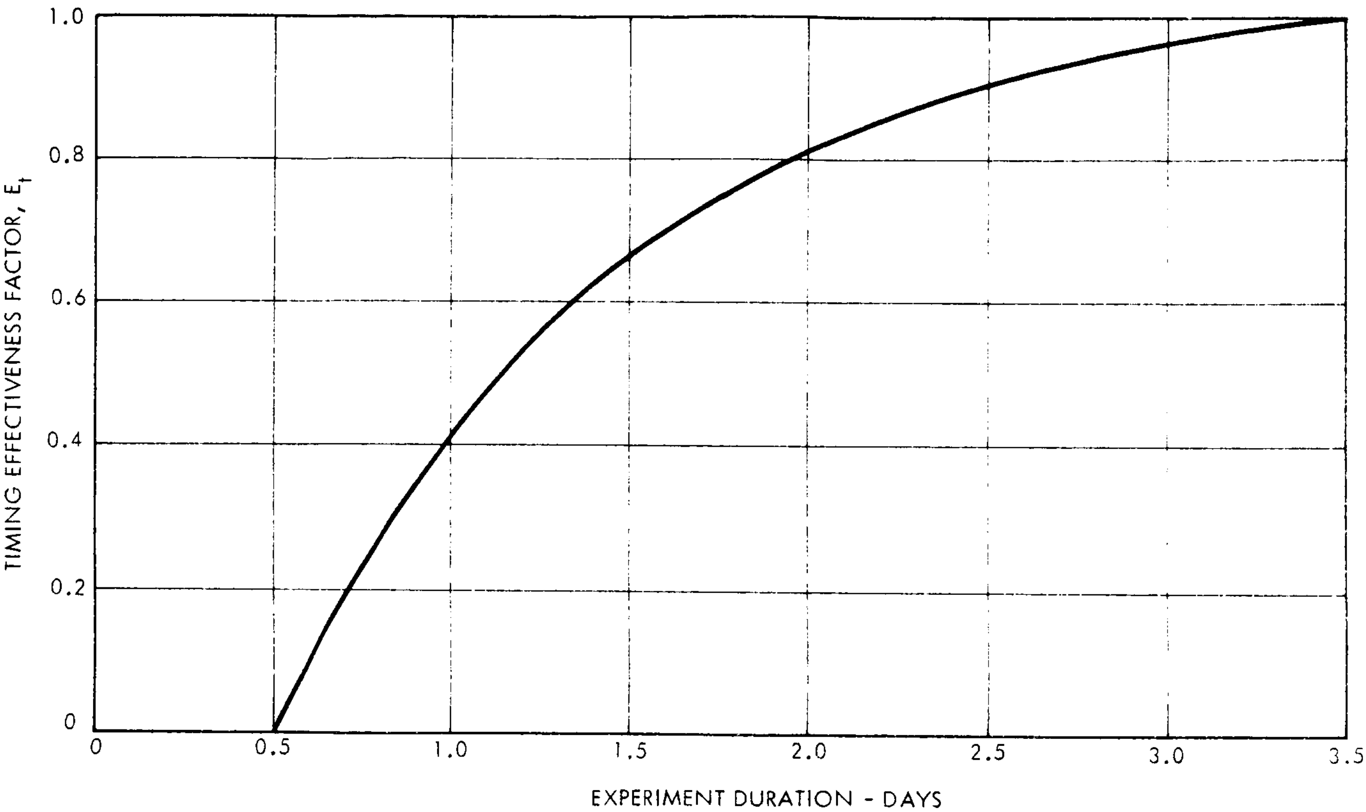

Figure B-42 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: M-2

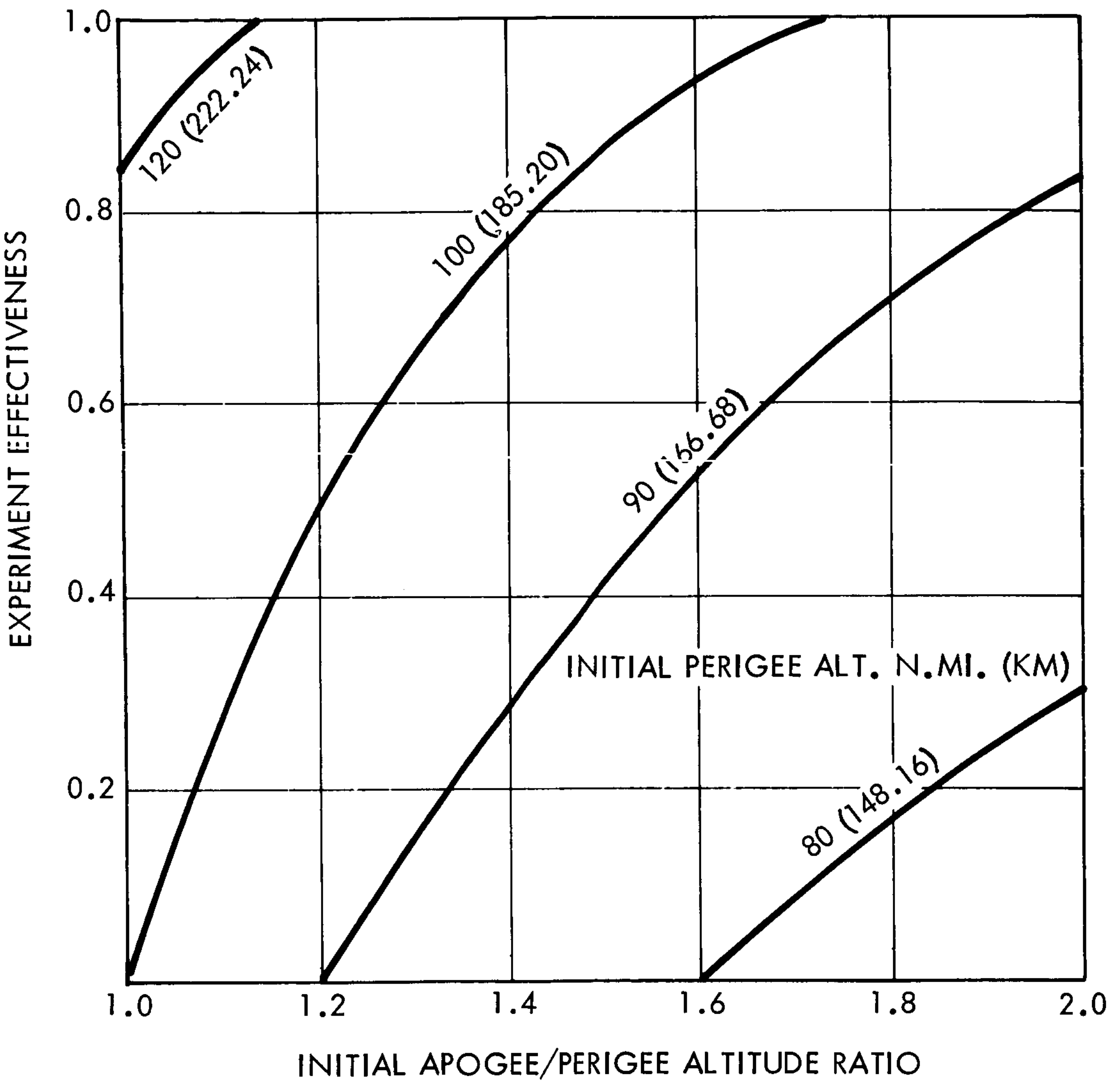

Figure B-43 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: M-2

Figure B-44 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: M-2

EXPERIMENT: M-2 **IBM PROBLEM NO.** 1439P012

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	1									
No. of Last Row	(IR)	8									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	7									

EFFECTIVENESS

PERIGEE ALT - km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	333.36	1.05	1.05	1.05	1.05	1.05	1.05								
2	259.28	1.05	1.05	1.05	1.05	1.05	1.05								
3	222.24	0.84	1.03	1.05	1.05	1.05	1.05								
4	185.20	0.02	0.49	0.75	0.94	1.01	1.04								
5	166.68	0.0	0.0	0.29	0.52	0.71	0.83								
6	148.16	0.0	0.0	0.0	0.0	0.17	0.30								
7	111.12	0.0	0.0	0.0	0.0	0.0	0.0								
8		1.0	1.2	1.4	1.6	1.8	2.0								
9		APOGEE/PERIGEE ALT. RATIO													
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

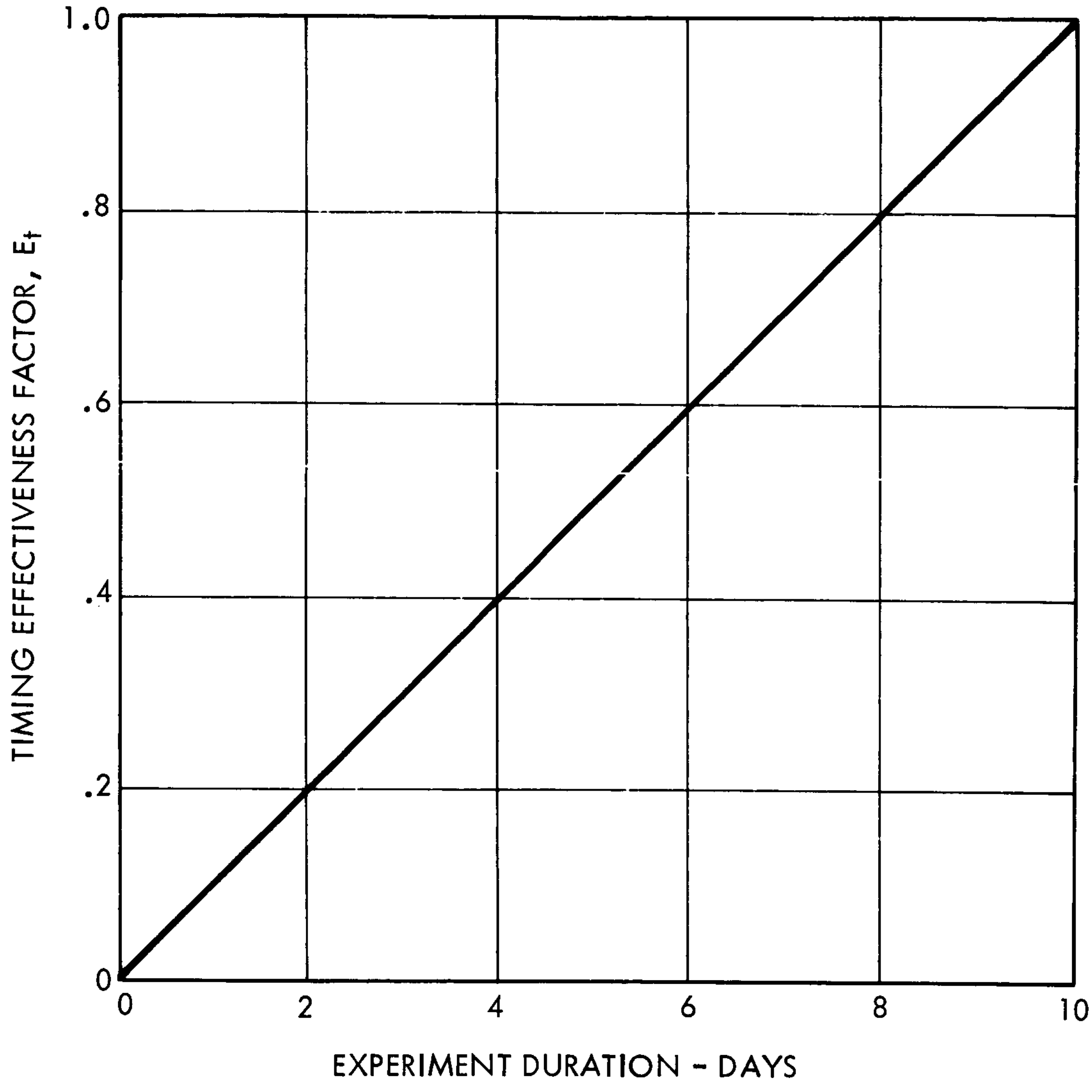

Figure B-45 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: M-3

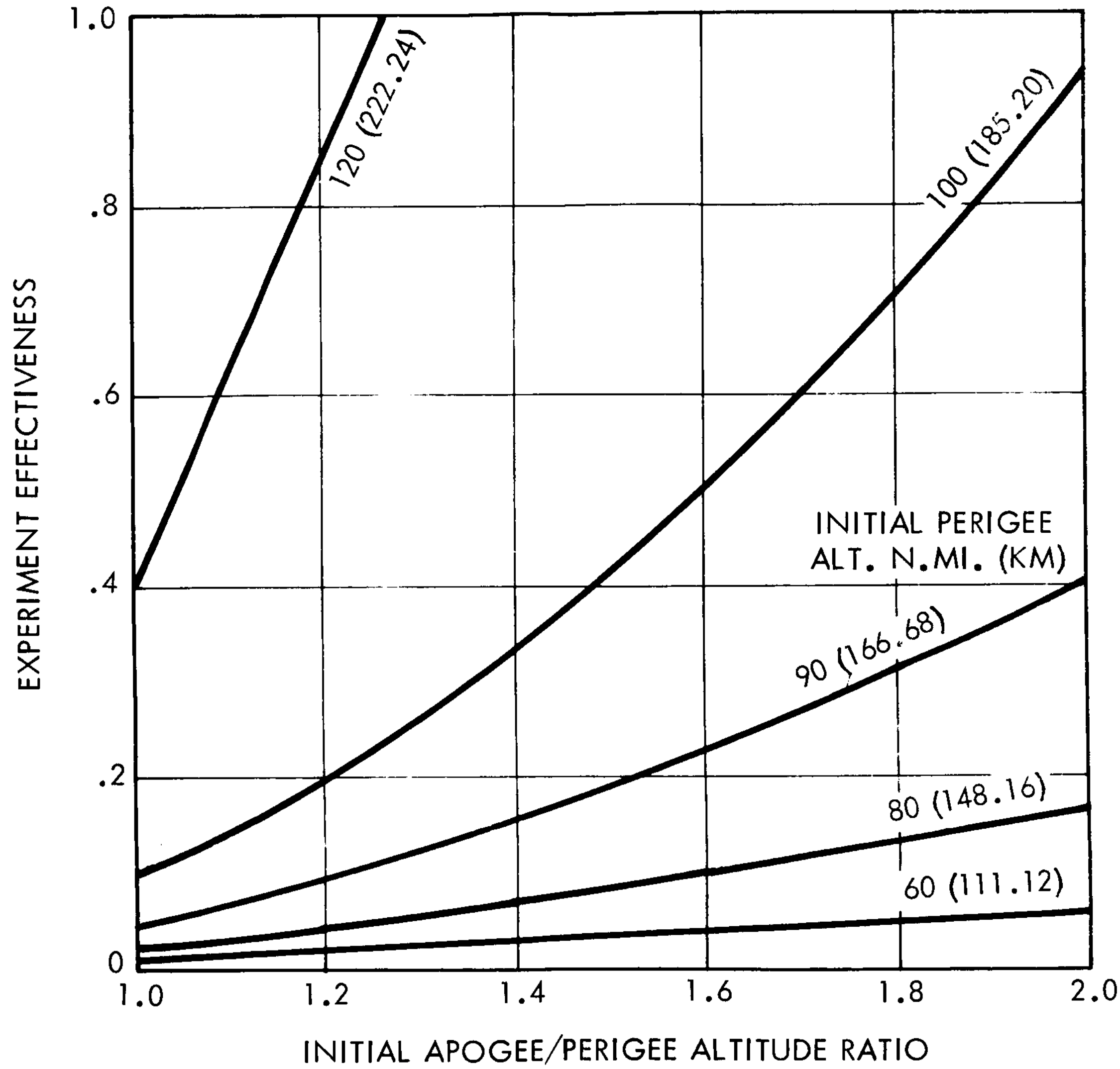

Figure B-46 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: M-3

Figure B-47 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: M-3

EXPERIMENT: M-3 **IBM PROBLEM NO.** 1439P013

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	2									
No. of Last Row	(IR)	9									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	8									

EFFECTIVENESS

PERIGEE ALT. - km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	333.36	5.18	9.30	16.1	25.3	38.0	53.7								
2	296.32	2.30	4.41	7.01	8.20	14.0	18.4								
3	259.28	1.07	2.13	3.07	4.58	6.08	7.76								
4	222.24	0.399	0.843	1.32	1.88	2.43	3.07								
5	185.20	0.100	0.217	0.339	0.518	0.713	0.939								
6	166.68	0.043	0.096	0.157	0.226	0.312	0.403								
7	148.16	0.022	0.041	0.067	0.096	0.130	0.159								
8	111.12	0.010	0.018	0.028	0.038	0.043	0.059								
9	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
10			APOGEE/PERIGEE ALT. RATIO												
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
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24															
25															

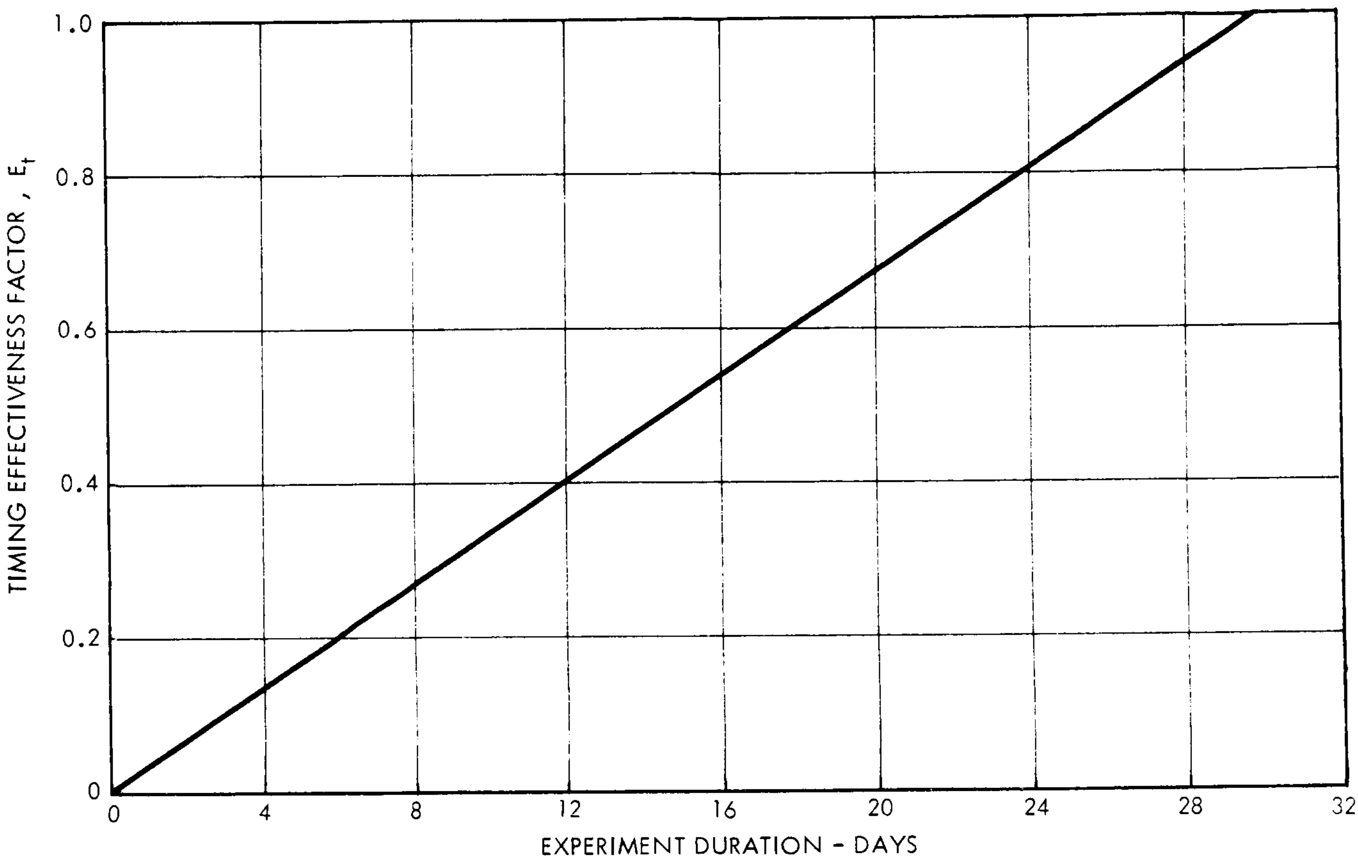

Figure B-48 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: M-4

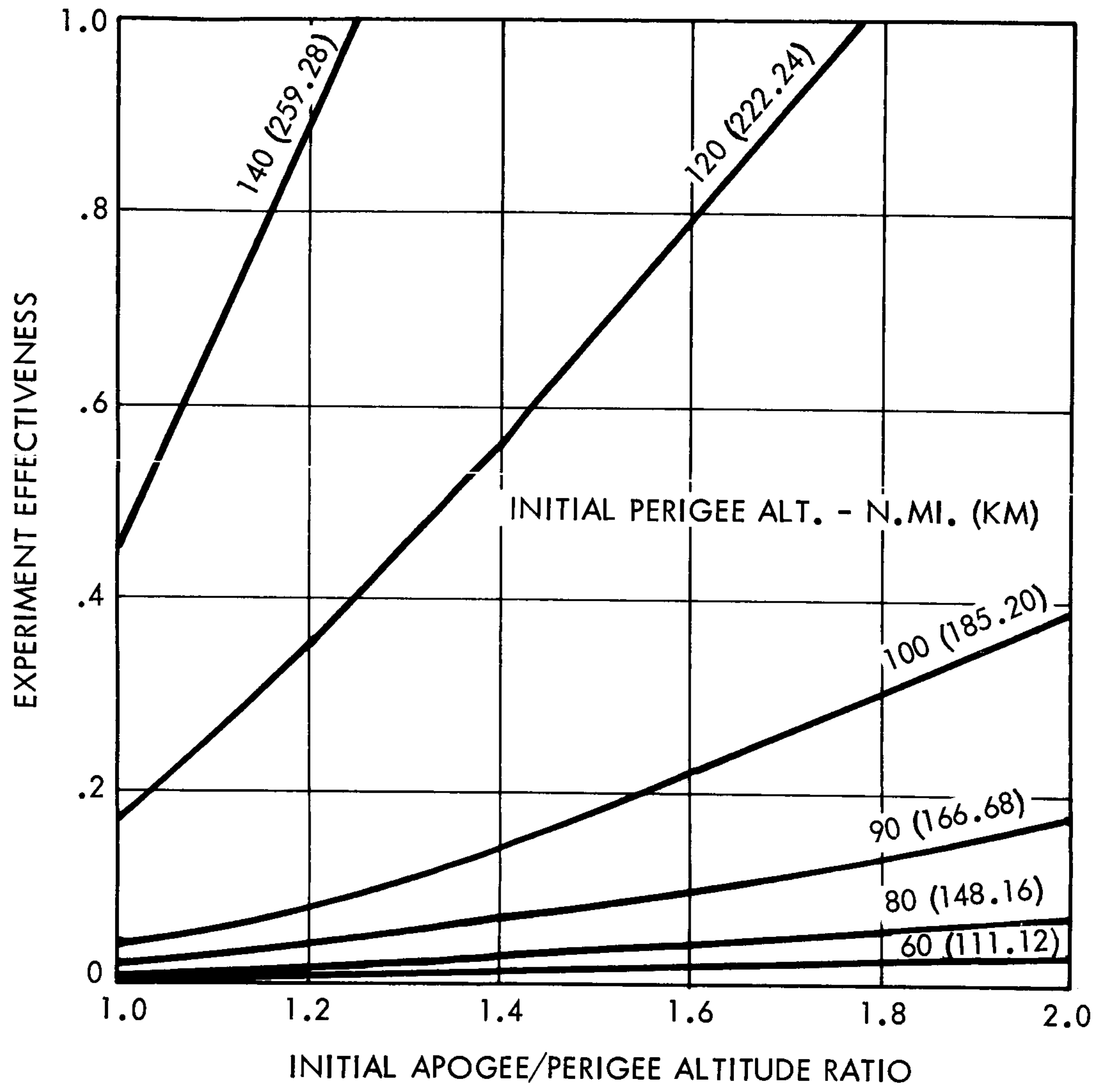

Figure B-49 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: M-4

Figure B-50 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: M-4

EXPERIMENT: M-4 **IBM PROBLEM NO.** 1439P014

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	2									
No. of Last Row	(IR)	9									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	8									

EFFECTIVENESS

PERIGEE ALT. - km

	1	2	3	4	5	6	7
1	333.36	2.18	3.91	6.77	10.6	16.0	22.6
2	296.32	0.97	1.85	2.98	3.45	5.88	7.73
3	259.28	0.45	0.89	1.37	1.93	2.55	4.07
4	222.24	0.17	0.35	0.56	0.79	1.02	1.29
5	185.20	0.042	0.091	0.14	0.22	0.30	0.39
6	166.68	0.02	0.04	0.066	0.095	0.13	0.17
7	148.16	0.009	0.017	0.028	0.040	0.055	0.067
8	111.12	0.004	0.007	0.012	0.016	0.020	0.025
9	0.0	1.0	1.2	1.4	1.6	1.8	2.0

APOGEE/PERIGEE ALT. RATIO

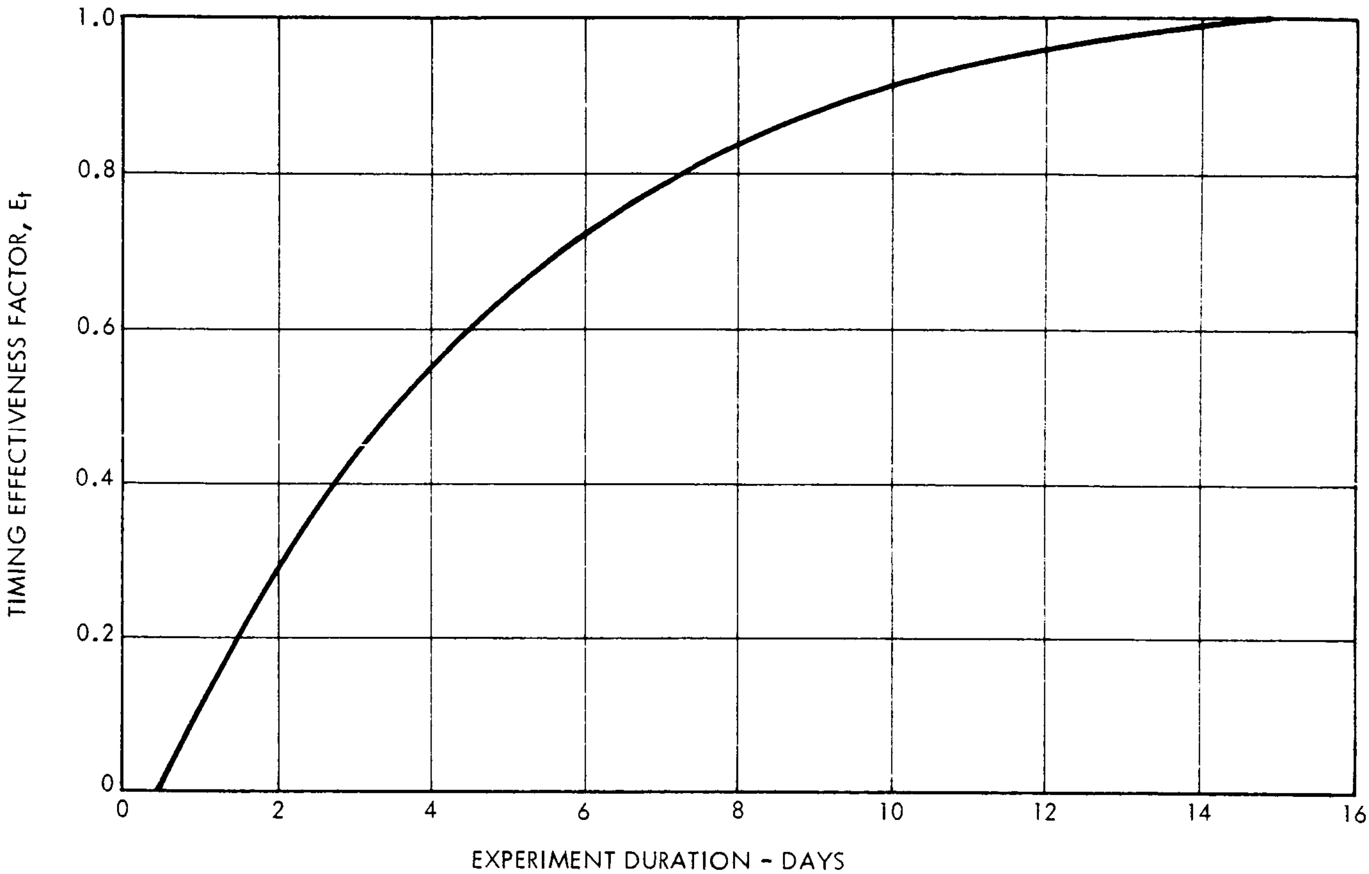

Figure B-51 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: M-5

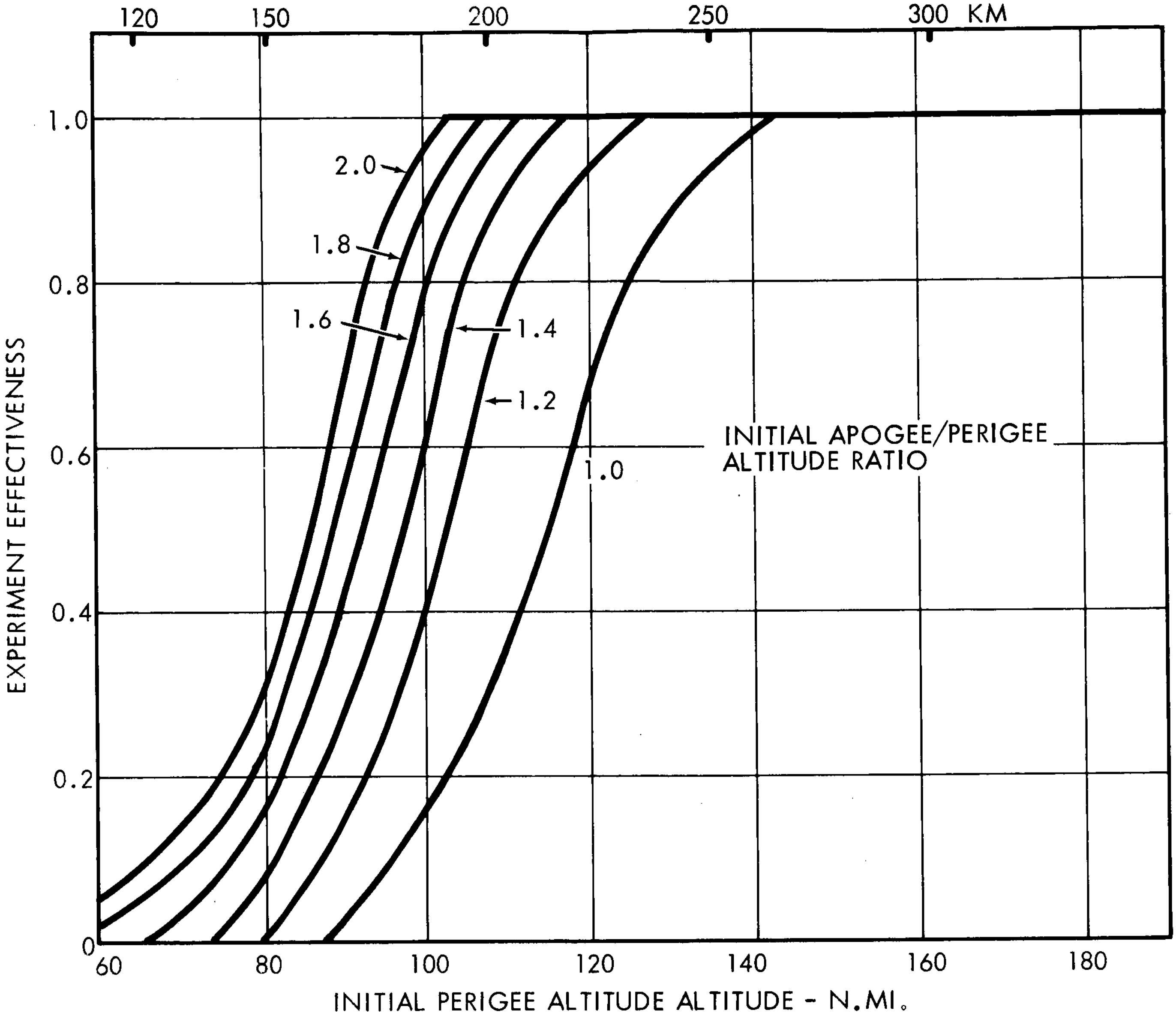

Figure B-52 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: M-5

Figure B-53 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: M-5

EXPERIMENT: M-5 **IBM PROBLEM NO.** 1439P015

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	10									
Second Variable I.D.	(KY)	11									
Interp. Option	(KI)	1									
No. of Last Row	(IR)	7									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	8									
No. of Ordinate Values	(NY)	6									

EFFECTIVENESS

APOGEE/PERIGEE ALT.	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	2.0	0.05	0.285	0.65	0.955	1.05	1.05	1.05	1.05						
2	1.8	0.02	0.22	0.54	0.88	1.05	1.05	1.05	1.05						
3	1.6	0.0	0.145	0.41	0.76	1.04	1.05	1.05	1.05						
4	1.4	0.0	0.07	0.28	0.58	1.02	1.05	1.05	1.05						
5	1.2	0.0	0.0	0.145	0.395	0.93	1.05	1.05	1.05						
6	1.0	0.0	0.0	0.02	0.15	0.65	0.98	1.05	1.05						
7	7.1	111.12	148.16	166.68	185.20	222.24	259.28	296.32	333.36						
8			PERIGEE ALT - km												
9															
10															
11															
12															
13															
14															
15															
16															
17															
18															
19															
20															
21															
22															
23															
24															
25															

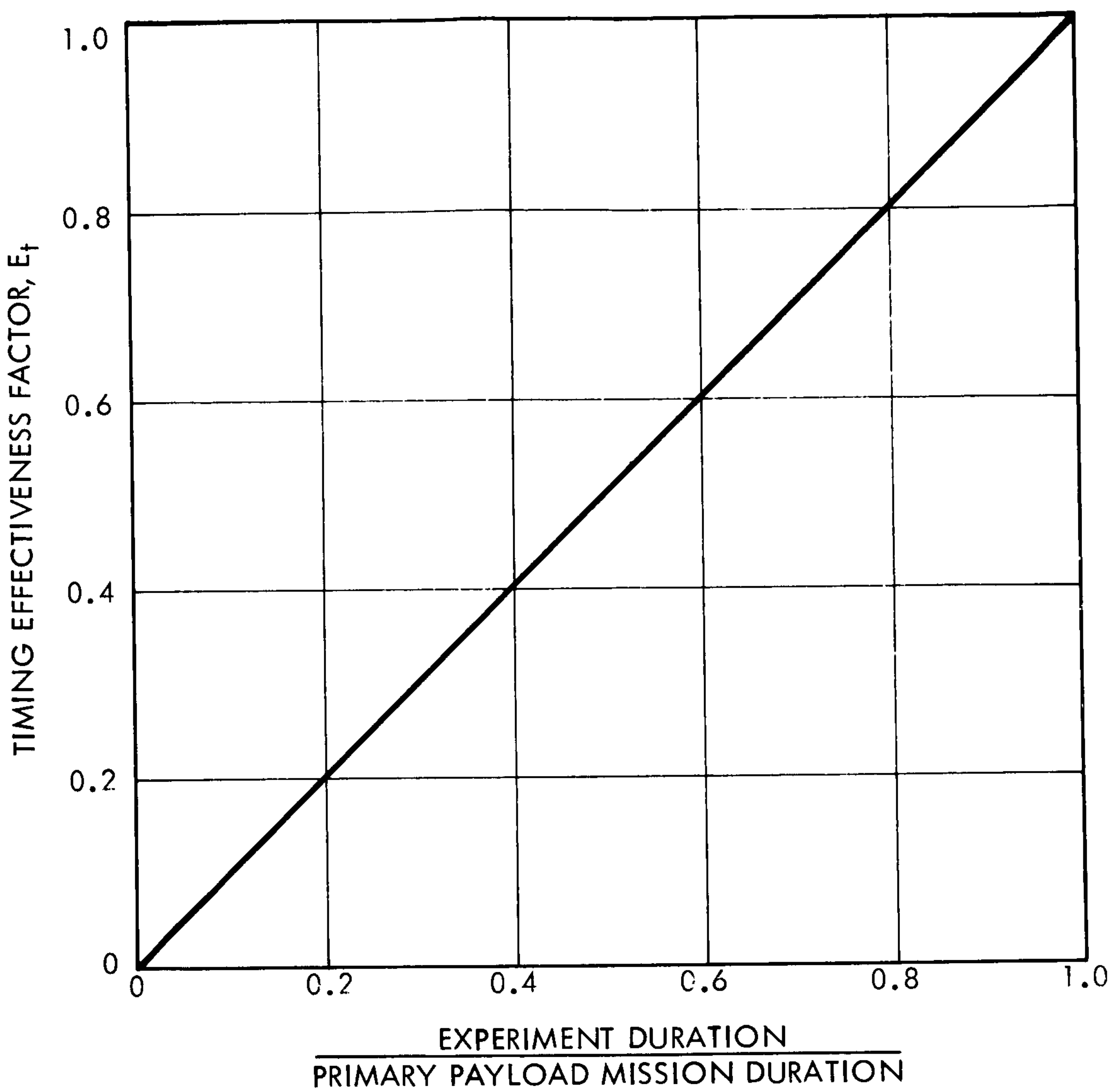

Figure B-54 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: OEA-1

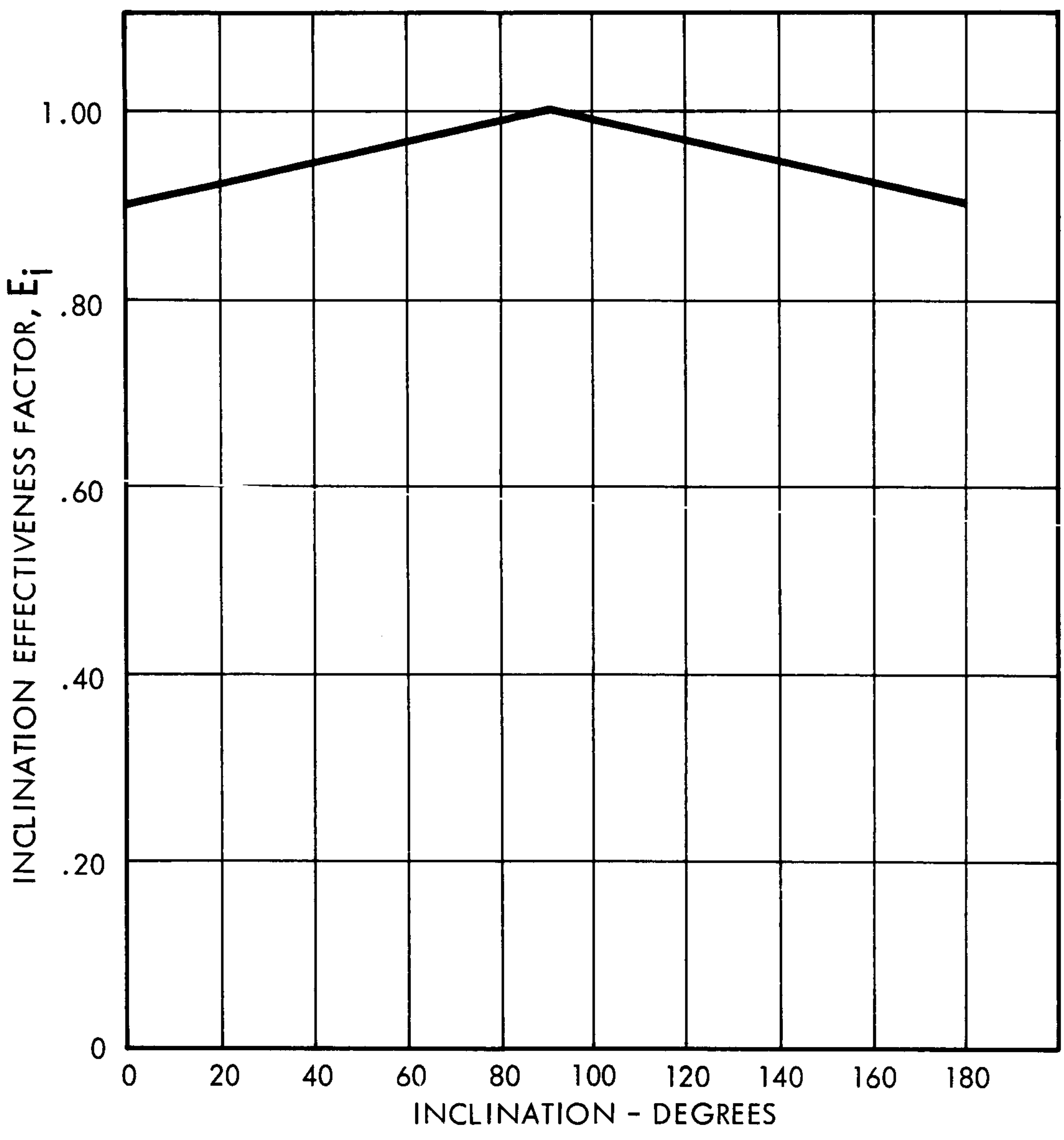

Figure B-55 FINAL EFFECTIVENESS DEFINITION - 1, EXPERIMENT: OEA-1

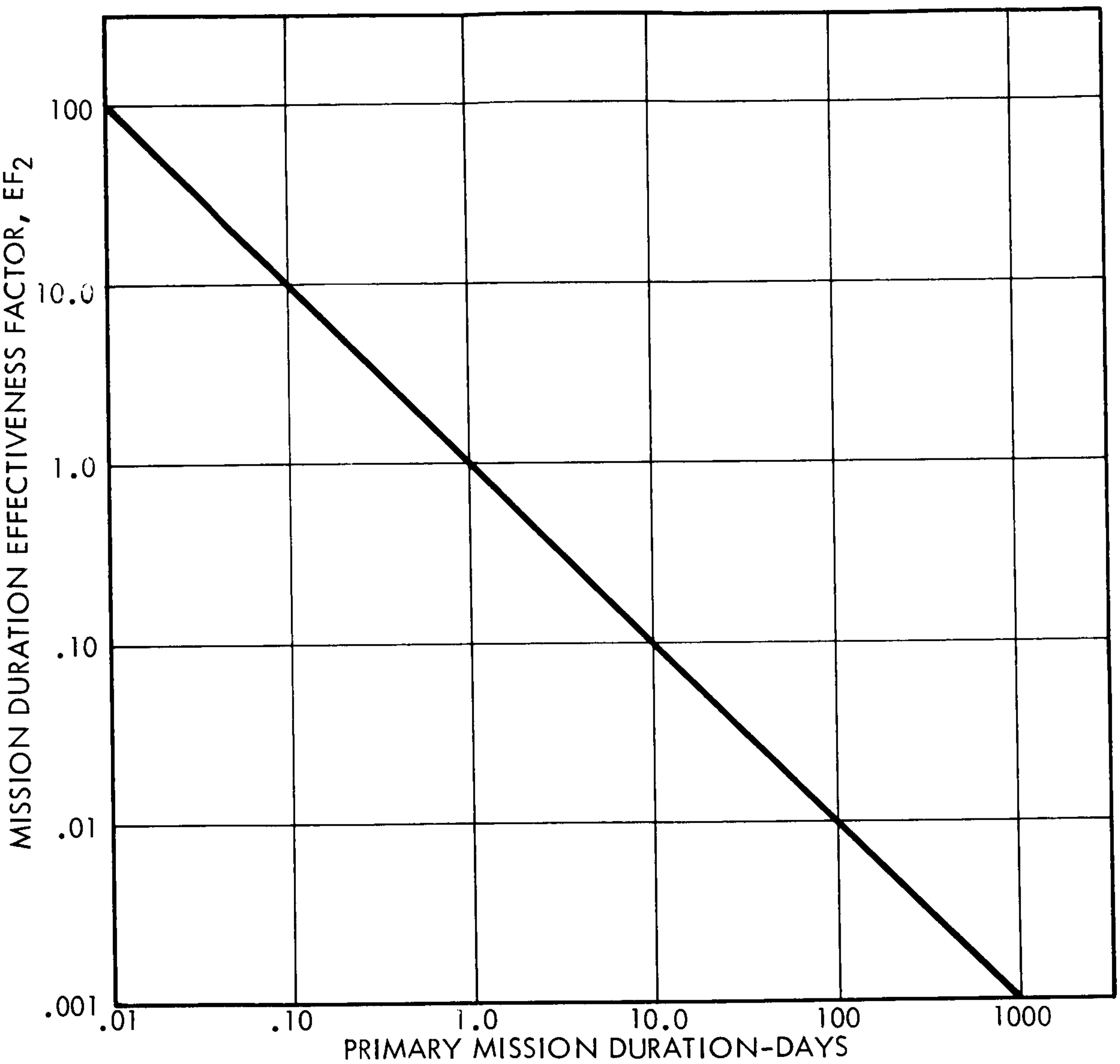

Figure B-56 FINAL EFFECTIVENESS DEFINITION - 2, EXPERIMENT: OEA-1

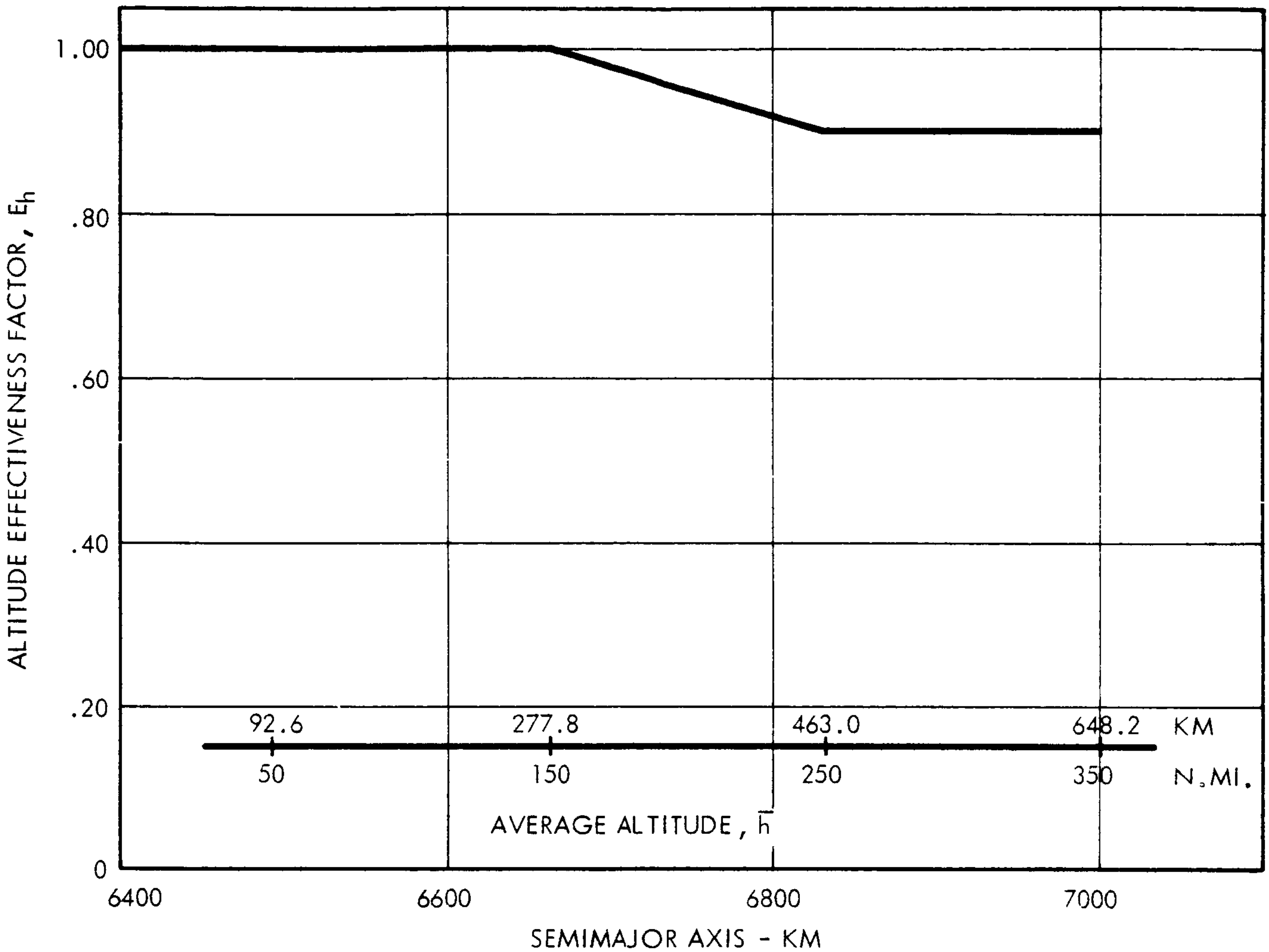

Figure B-57 FINAL EFFECTIVENESS DEFINITION - 3, EXPERIMENT: OEA-1

Figure B-58 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: OEA-1

EXPERIMENT: OEA-1 **IBM PROBLEM NO.** 1439P026

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	10	19	1	3						
Second Variable I.D.	(KY)	11	0	0	0						
Interp. Option	(KI)	2	1	1	1						
No. of Last Row	(IR)	7	11	14	17						
No. of First Column	(JC)	1	1	1	1						
No. of Abscissa Values	(NX)	8	14	4	3						
No. of Ordinate Values	(NY)	6	1	1	1						

APOGEE/PERIGEE ALT. RATIO

EFFECTIVENESS

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	2.0	0.31	0.83	2.10	4.90	16.00	40.50	96.0	280.0						
2	1.8	0.25	0.68	1.63	3.72	12.70	31.7	73.0	198.0						
3	1.6	0.20	0.50	1.18	2.70	9.80	23.9	42.8	132.0						
4	1.4	0.15	0.35	0.82	1.77	6.90	17.0	37.0	84.0						
5	1.2	0.09	0.22	0.50	1.13	4.40	11.1	23.0	48.5						
6	1.0	0.05	0.12	0.25	0.52	2.08	5.60	12.0	27.0						
7	0.0	111.12	148.16	166.68	185.20	222.24	259.28	296.32	333.36	INITIAL PERIGEE ALT.					
8															
9						EFFECTIVENESS									
10	0.0	1000.0	100.0	10.0	5.0	2.0	1.5	1.25	1.0	0.8	.666667	0.5	0.1	0.01	0.001
11	0.0	0.001	0.01	0.1	0.2	0.5	.666667	0.8	1.0	1.25	1.5	2.0	10.0	100.0	1000.0
12			EFFECTIVENESS								MISSION DURATION				
13	0.0	1.0	1.0	0.9	0.9										
14	0.0	0.0	6648.68	6833.88	6833.88	SEMI-MAJOR AXIS									
15															
16	0.0	0.9	1.0	0.9											
17	0.0	0.0	90.0	180.0	INCLINATION										
18															
19															
20															
21															
22															
23															
24															
25															

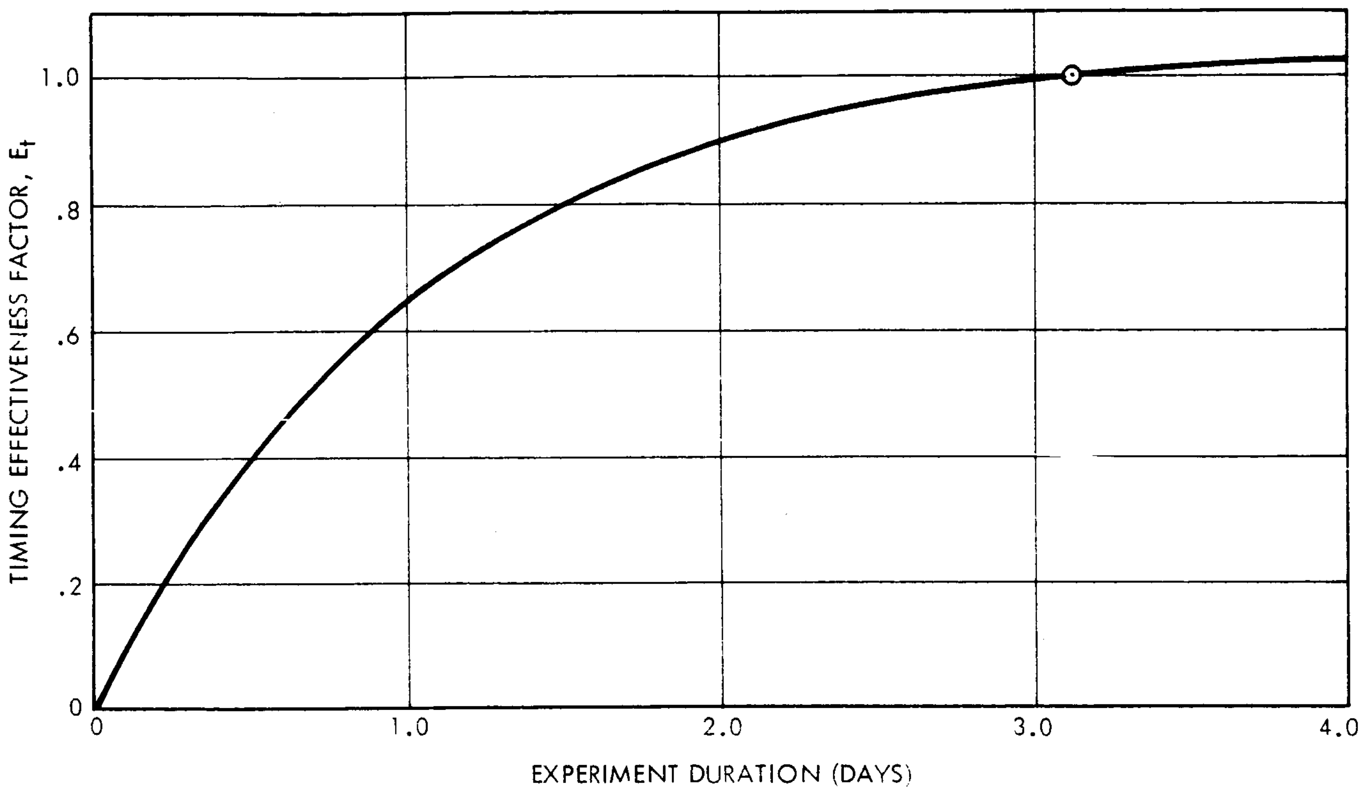

Figure B-59 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: OEA-2

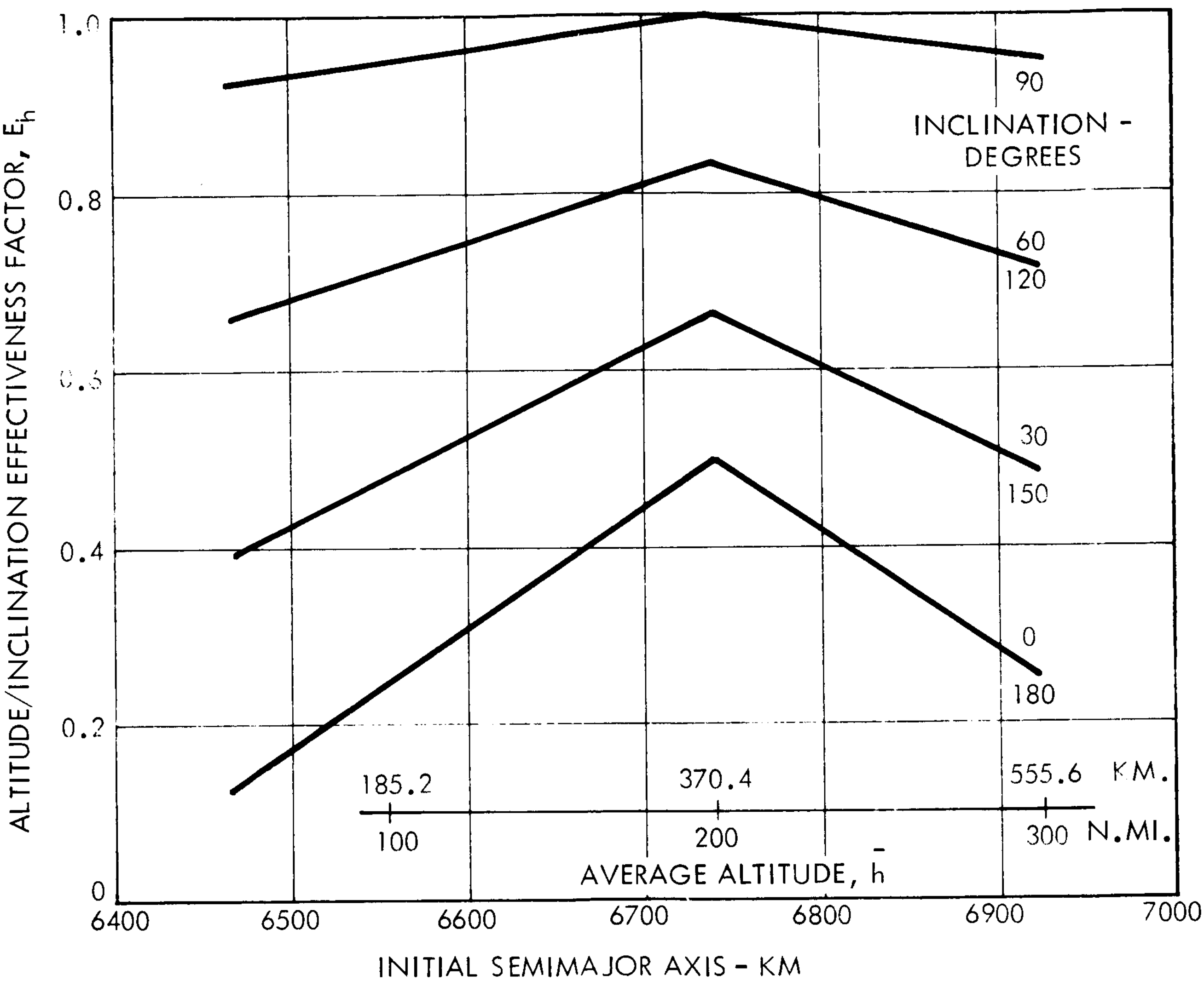

Figure B-60 FINAL EFFECTIVENESS DEFINITION - 1, EXPERIMENT: OEA-2

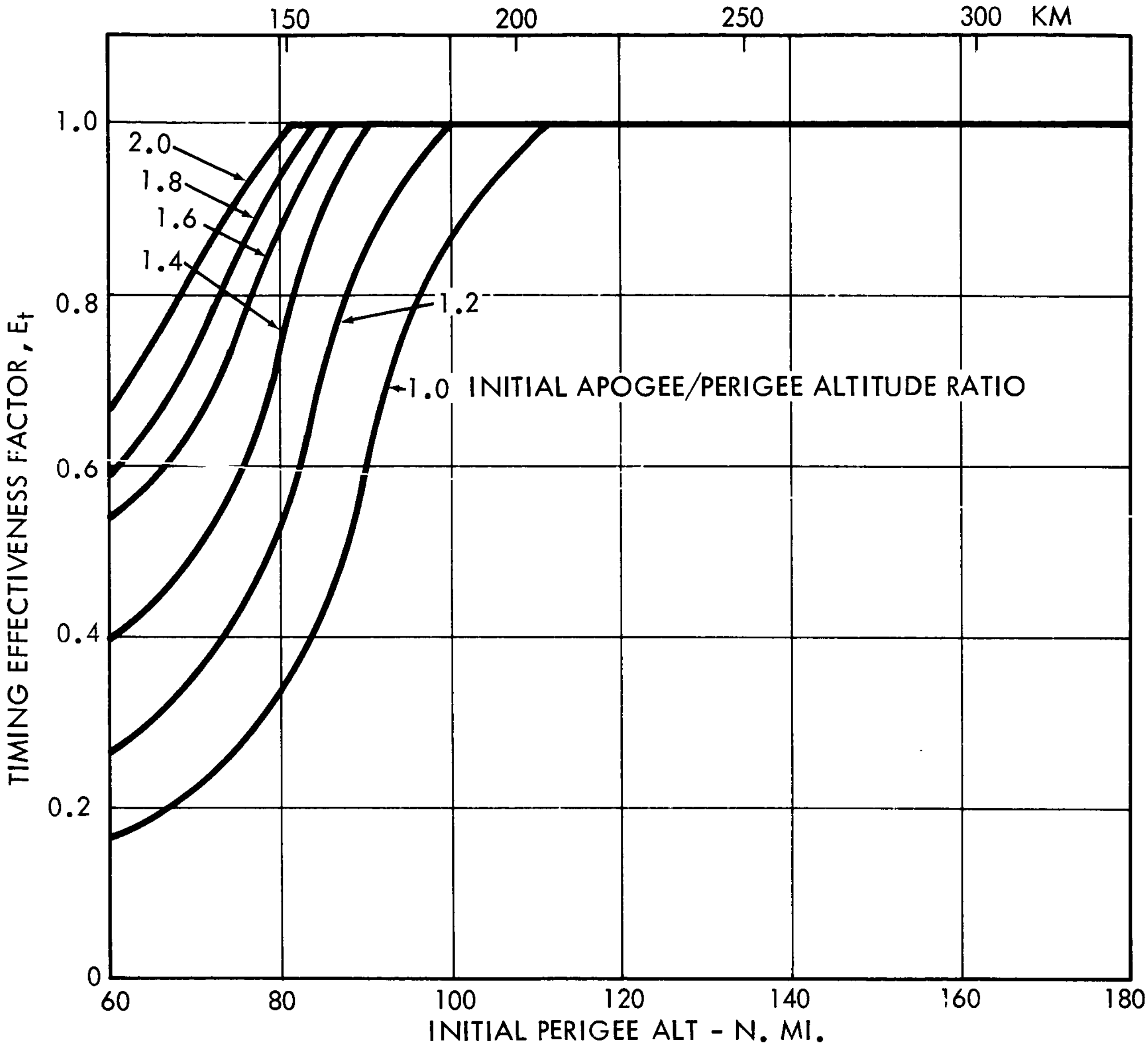

Figure B-61 FINAL EFFECTIVENESS DEFINITION - 2, EXPERIMENT: OEA-2

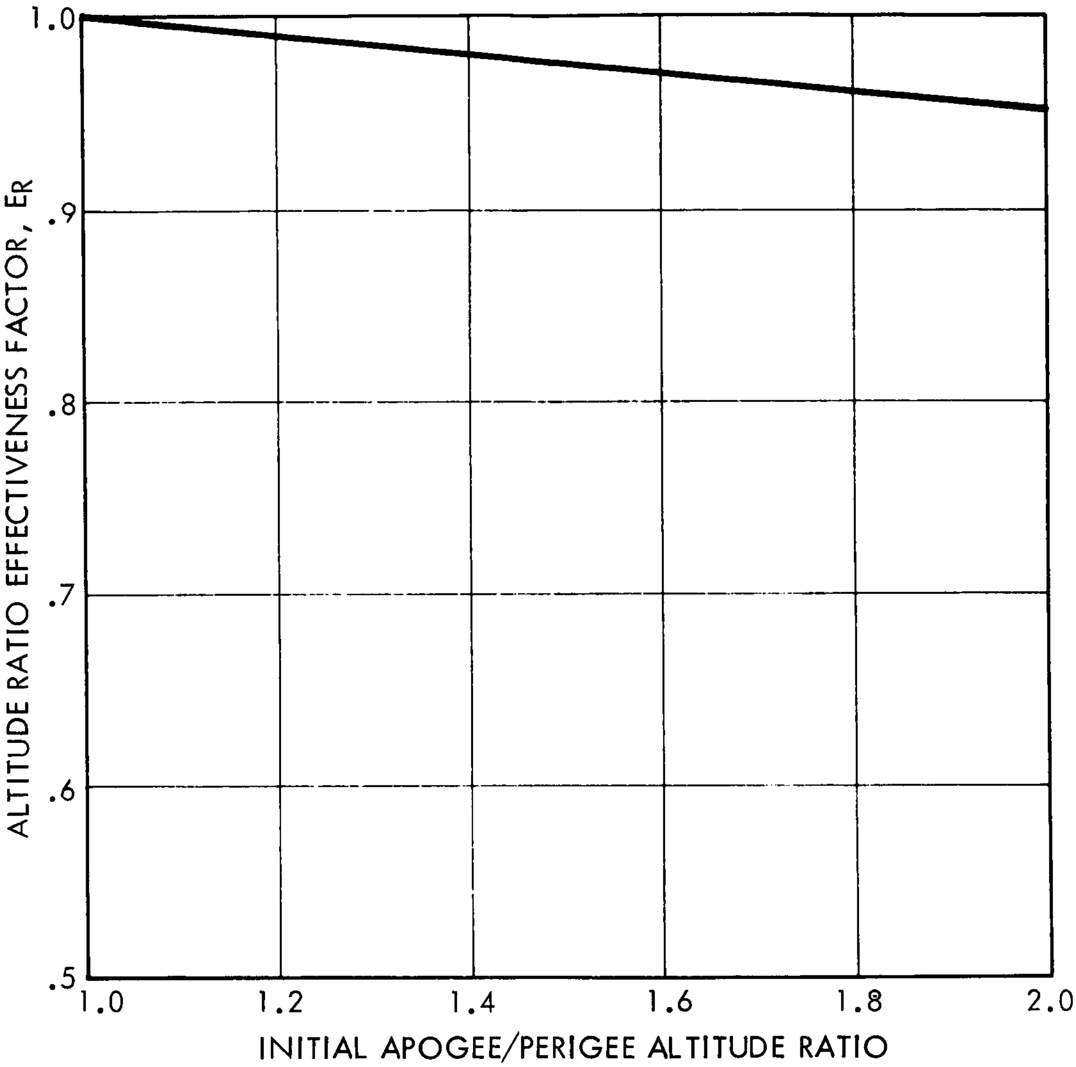

Figure B-62 FINAL EFFECTIVENESS DEFINITION - 3, EXPERIMENT: OEA-2

Figure B-63 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: OEA-2

EXPERIMENT: OEA-2 **IBM PROBLEM NO.** 1439P027

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	1	10	11							
Second Variable I.D.	(KY)	3	11	0							
Interp. Option	(KI)	2	1	1							
No. of Last Row	(IR)	8	17	21							
No. of First Column	(JC)	1	1	1							
No. of Abscissa Values	(NX)	6	6	2							
No. of Ordinate Values	(NY)	7	6	1							

EFFECTIVENESS

	Row	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
INCLINATION - deg	1	180.0	0.125	0.250	0.375	0.500	0.375	0.250								
	2	150.0	0.392	0.483	0.575	0.667	0.575	0.483								
	3	120.0	0.658	0.716	0.775	0.834	0.775	0.716								
	4	90.0	0.925	0.950	0.975	1.0	0.975	0.950								
	5	60.0	0.658	0.716	0.775	0.834	0.775	0.716								
	6	30.0	0.392	0.483	0.575	0.667	0.575	0.483								
	7	0.0	0.125	0.250	0.375	0.500	0.375	0.250								
	8	0.0	6463.48	6556.08	6648.68	6741.28	6833.88	6926.48	SEMI-MAJOR AXIS - km							
	9															
	10															
APOGEE/PERIGEE ALT. RATIO	11	2.0	0.670	0.981	1.05	1.05	1.05	1.05								
	12	1.8	0.59	0.94	1.04	1.05	1.05	1.05								
	13	1.6	0.54	0.848	1.03	1.05	1.05	1.05								
	14	1.4	0.395	0.712	0.98	1.05	1.05	1.05								
	15	1.2	0.265	0.532	0.848	1.02	1.05	1.05								
	16	1.0	0.16	0.325	0.59	0.86	1.05	1.05								
	17	0.0	111.12	148.16	166.68	185.2	222.24	333.36	INITIAL PERIGEE ALT - km							
	18															
	19															
	20	0.0	1.00	0.95												
	21	0.0	1.0	2.0	APOGEE/PERIGEE ALT. RATIO											
	22															
	23															
	24															
	25															

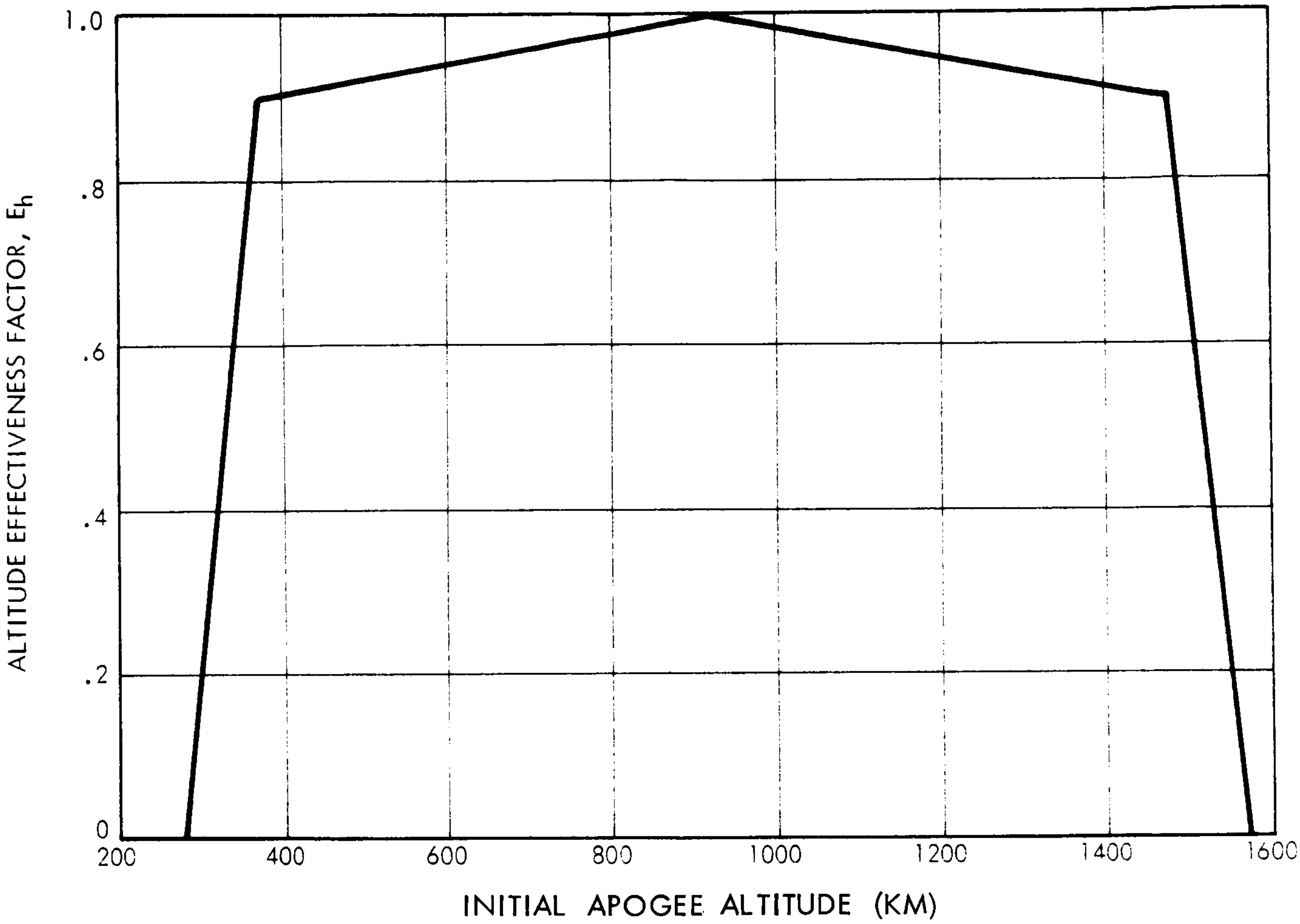

Figure B-64 FINAL EFFECTIVENESS DEFINITION - 1, EXPERIMENT: OEA-3

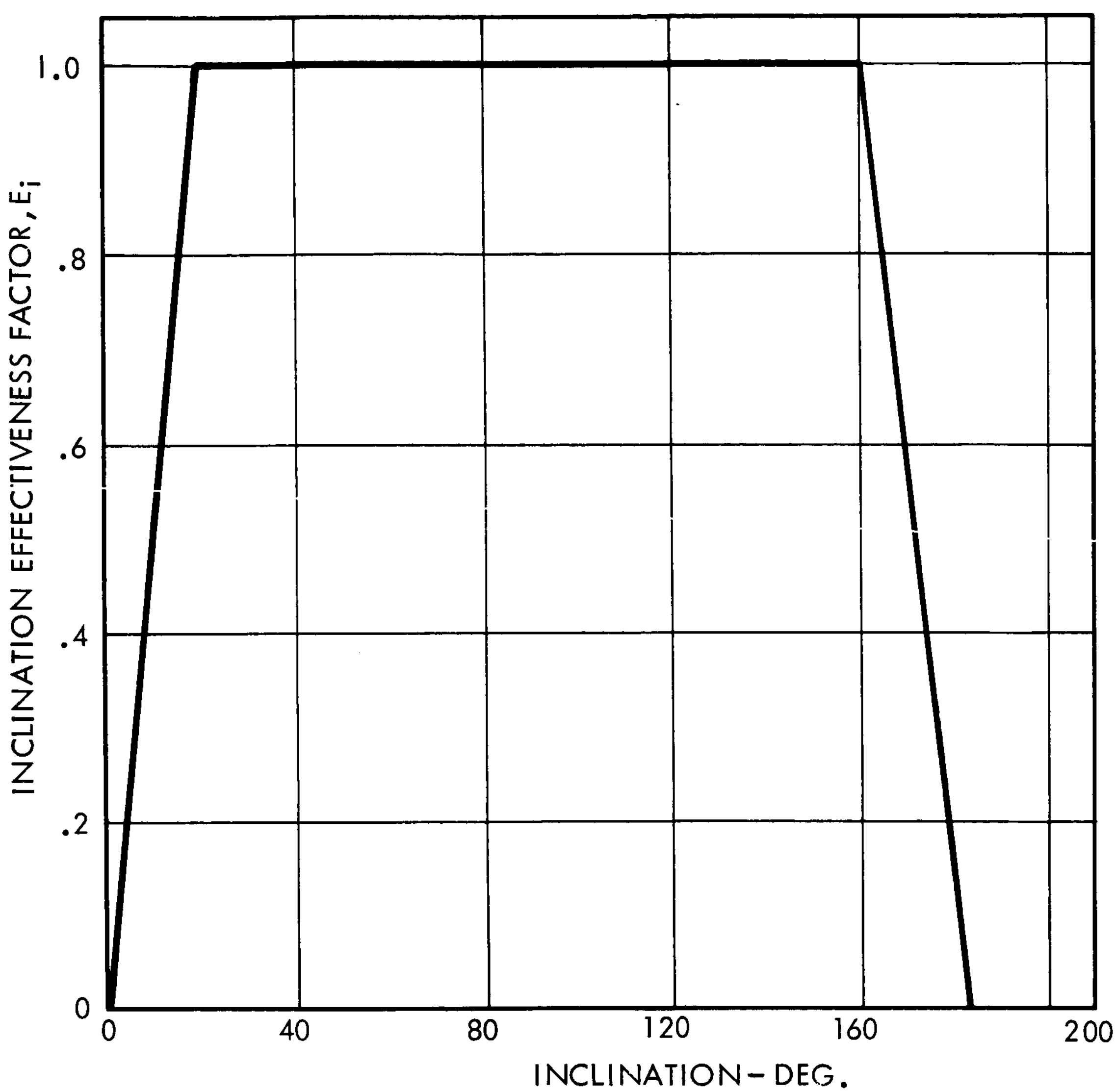

Figure B-65 FINAL EFFECTIVENESS DEFINITION - 2, EXPERIMENT: OEA-3

Figure B-66 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: OEA-3

EXPERIMENT: OEA-3 **IBM PROBLEM NO.** 1439P028

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	9	3								
Second Variable I.D.	(KY)	0	0								
Interp. Option	(KI)	1	1								
No. of Last Row	(IR)	2	5								
No. of First Column	(JC)	1	1								
No. of Abscissa Values	(NX)	5	4								
No. of Ordinate Values	(NY)	1	1								

	1	2	3	4	5	6	
	TABLE NO. 1			EFFECTIVENESS			
1	0.0	0.0	0.9	1.0	0.9	0.0	
2	0.0	277.8	370.4	296.0	1481.6	1574.2	APOGEE ALT. - km
3	TABLE NO. 2			EFFECTIVENESS			
4	0.0	0.0	1.0	1.0	0.0		
5	0.0	0.0	20.0	160.0	180.0		INCLINATION - deg
6							
7							
8							
9							
10							
11							
12							
13							
14							
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17							
18							
19							
20							
21							
22							
23							
24							
25							

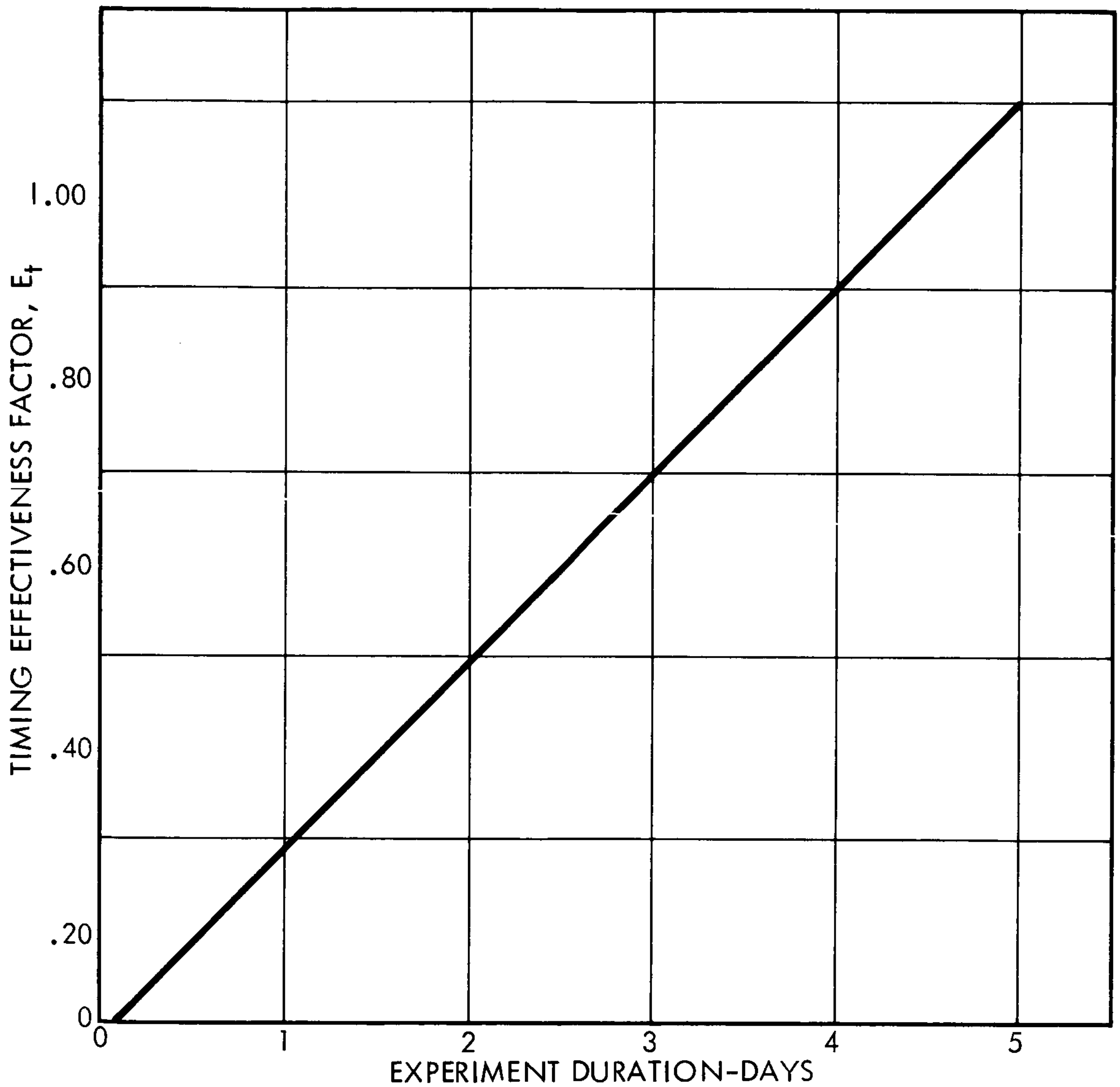

Figure B-67 BASIC EFFECTIVENESS DEFINITION, EXPERIMENT: OEA-4

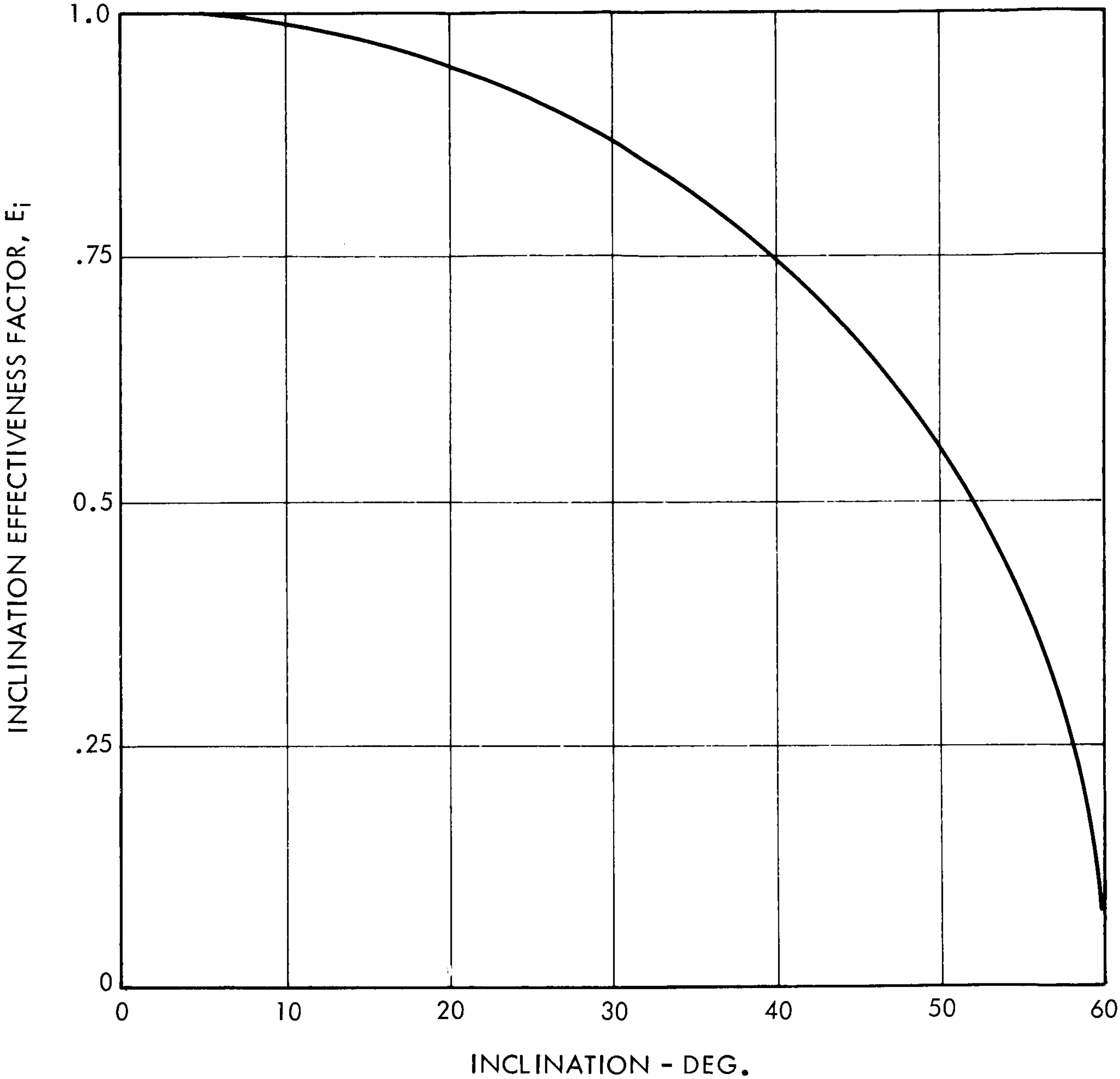

Figure B-68 FINAL EFFECTIVENESS DEFINITION - 1, EXPERIMENT: OEA-4

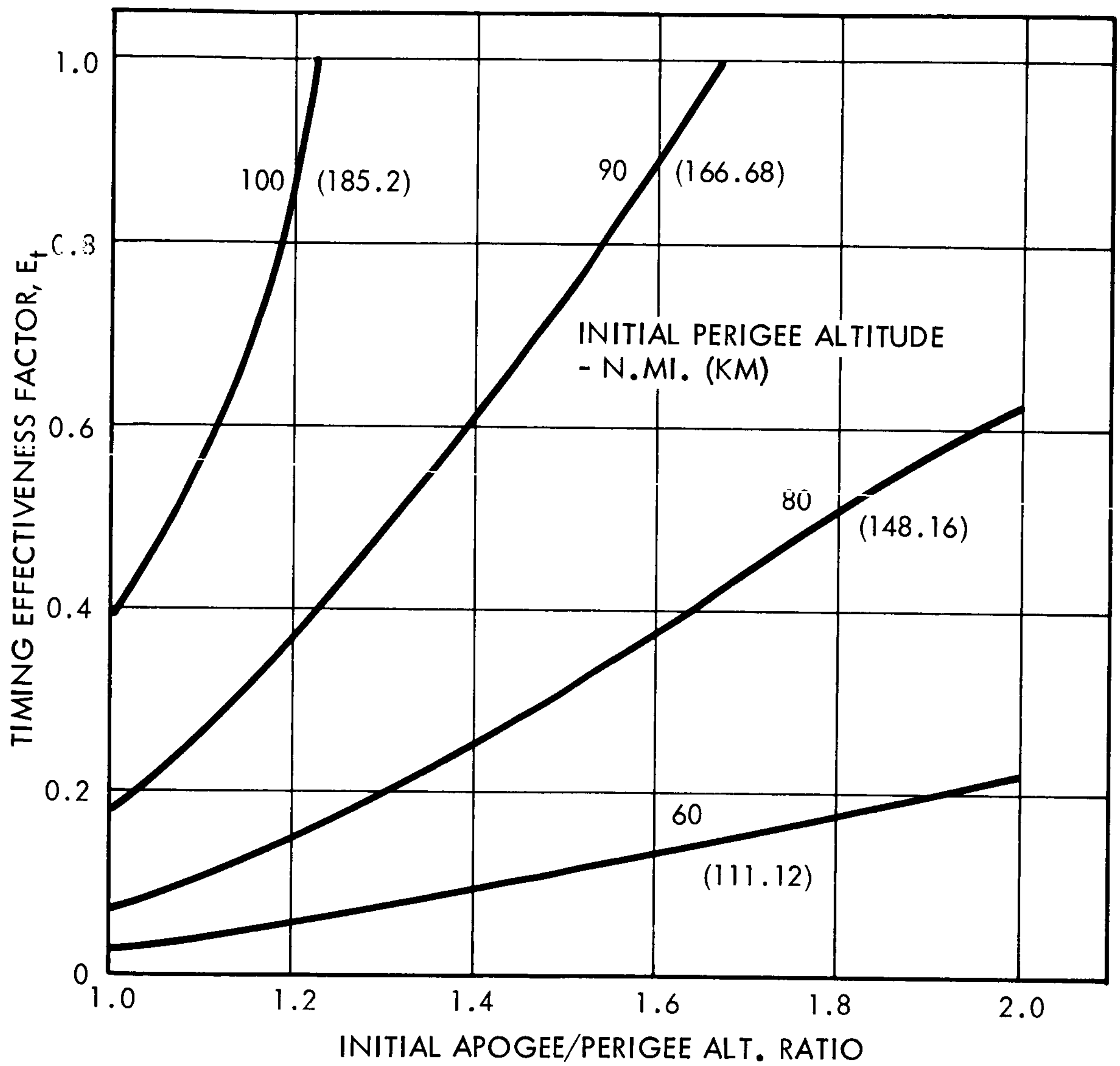

Figure B-69 FINAL EFFECTIVENESS DEFINITION - 2, EXPERIMENT: OEA-4

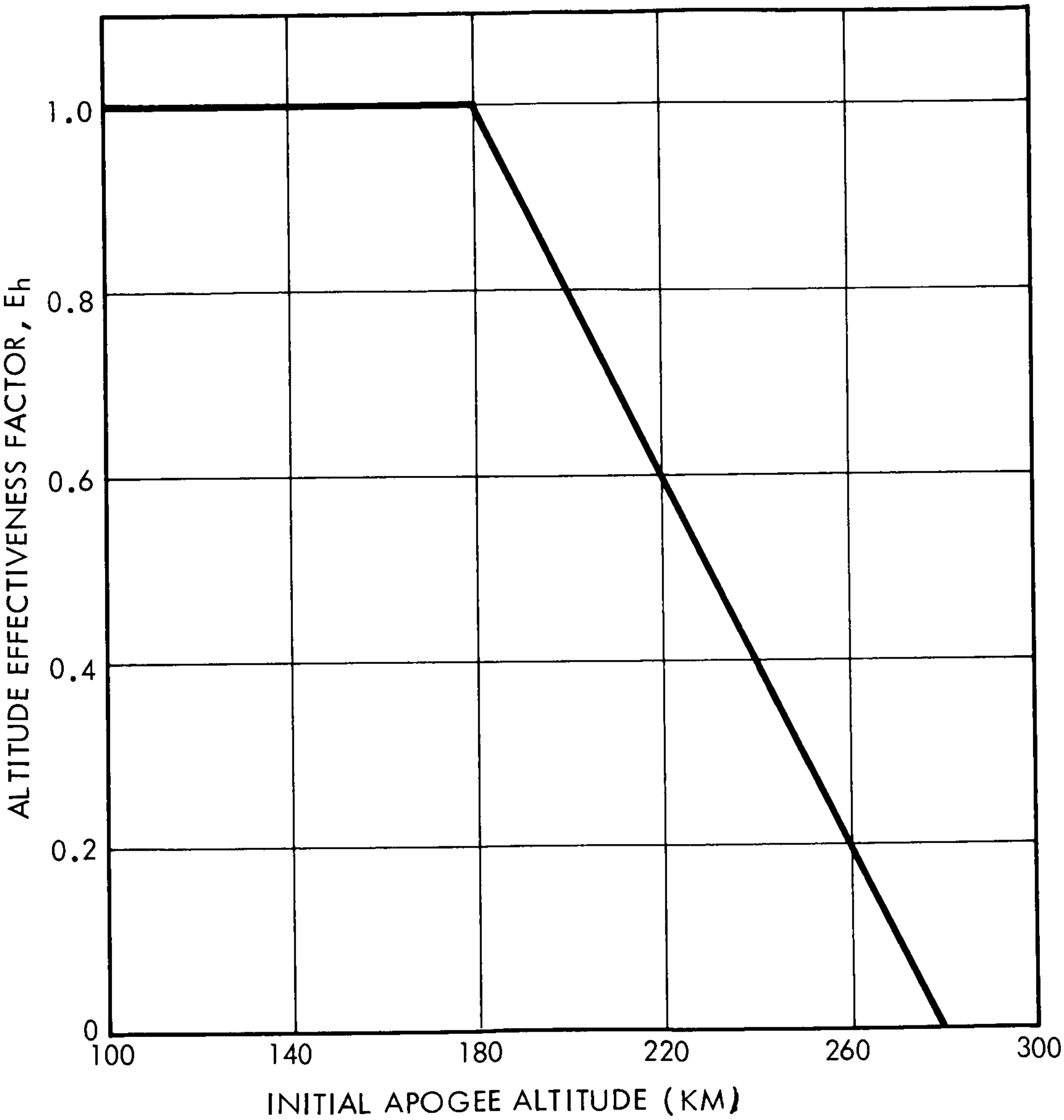

Figure B-70 FINAL EFFECTIVENESS DEFINITION - 3, EXPERIMENT: OEA-4

Figure B-71 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT OEA-4

EXPERIMENT: OEA-4 **IBM PROBLEM NO.** '1439P029

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	3	11	9							
Second Variable I.D.	(KY)	0	10	0							
Interp. Option	(KI)	1	1	1							
No. of Last Row	(IR)	2	11	15							
No. of First Column	(JC)	1	1	1							
No. of Abscissa Values	(NX)	14	6	4							
No. of Ordinate Values	(NY)	1	6	1							

PERIGEE ALT - km

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
	TABLE NO. 1					EFFECTIVENESS									
1	0.0	1.0	0.987	0.972	0.937	0.885	0.823	0.748	0.663	0.568	0.463	0.355	0.182	0.0	0.0
2	0.0	0.0	7.3	14.4	21.3	27.9	34.1	39.8	44.9	49.4	53.1	56.1	59.	60.	120.
3						INCLINATION - deg									
4	TABLE NO. 2			EFFECTIVENESS											
5	111.12	.0238	.05431	.09499	.13567	.17635	.22110								
6	148.16	.07262	.14787	.25161	.36755	.50586	.62180								
7	166.68	.17635	.36755	.61366	.89028	1.2381	1.5981								
8	185.20	.38382	.85367	1.3459	2.0598	2.84496	3.75416								
9	222.24	1.58388	3.3697	5.2919	7.5252	1.0	1.0								
10	333.36	1.0	1.0	1.0	1.0	1.0	1.0								
11	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
12			APOGEE/PERIGEE ALT. RATIO												
13	TABLE NO. 3		EFFECTIVENESS												
14	0.0	1.0	1.0	0.0	0.0										
15	0.0	100.0	180.0	280.0	100.0										
16			APOGEE ALT. - km												
17															
18															
19															
20															
21															
22															
23															
24															
25															

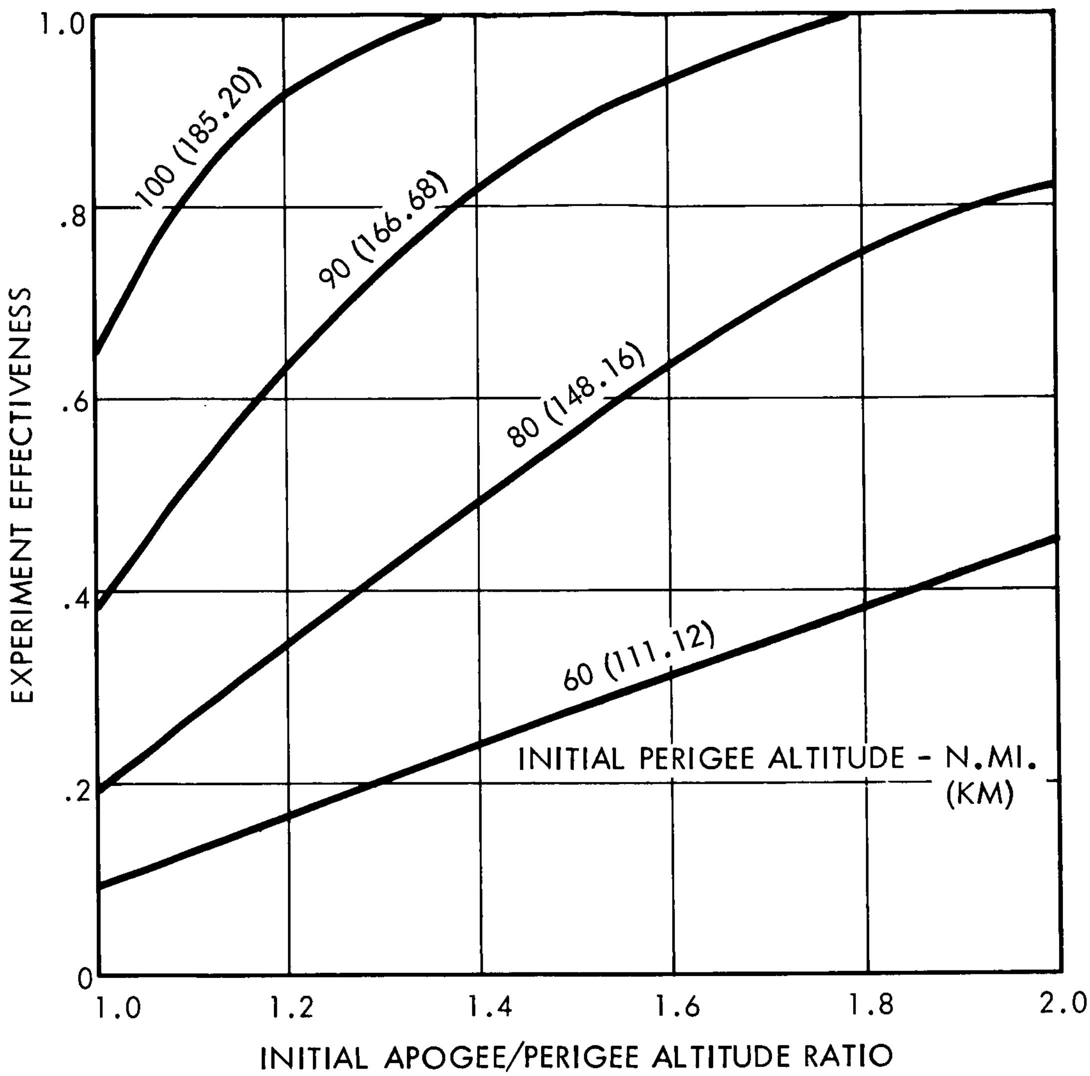

Figure B-72 FINAL EFFECTIVENESS DEFINITION, EXPERIMENT: OEA-5

Figure B-73 EXPERIMENT EFFECTIVENESS LIBRARY WORK SHEET, EXPERIMENT: OEA-5

EXPERIMENT: OEA-5 **IBM PROBLEM NO.** 143P030

		TABLE NO. 1	2	3	4	5	6	7	8	9	10
Abscissa Variable I.D.	(KX)	11									
Second Variable I.D.	(KY)	10									
Interp. Option	(KI)	1									
No. of Last Row	(IR)	7									
No. of First Column	(JC)	1									
No. of Abscissa Values	(NX)	6									
No. of Ordinate Values	(NY)	6									

INITIAL PERIGEE ALT.

	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	333.36	1.05	1.05	1.05	1.05	1.05	1.05								
2	222.24	1.03	1.05	1.05	1.05	1.05	1.05								
3	185.2	0.64	0.92	1.01	1.04	1.05	1.05								
4	166.68	0.38	0.63	0.82	0.93	1.00	1.02								
5	148.16	0.20	0.34	0.49	0.63	0.75	0.82								
6	111.12	0.09	0.16	0.24	0.31	0.38	0.45								
7	0.0	1.0	1.2	1.4	1.6	1.8	2.0								
8			APOGEE/PERIGEE ALT. RATIO												
9															
10															
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